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NOZZLE DESIGN CRITERIA FOR A MACH 4 LOW-DENSITY FACILITY USING CARBON DIOXIDE – WITH COMMENTS ON MACH 3, 6, 9 AND 12 NOZZLES

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Prepared for George C. Marshall Space Flight Center, Huntsville, Alabama under Contract NAS8-11144

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SUMMARY

This is the summary report for the Low-Density Wind Tunnel Nozzle Design Criteria, Contract NAS8-11144. This work was performed in support of the Aerodynamics Division of the Marshall Space Flight Center Aero-Astrodynamics Laboratory. A summary of the research work is presented along with the design criteria for nozzles sized to fit the MSFC vacuum chamber having exit Mach numbers of 3, 4, 6, 9 and 12.

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The University of Southern California Engineering Center (USCEC), prior to this contract, conducted a low-density wind tunnel nozzle design criteria study for similar Mach numbers but used nitrogen as a working gas for the nozzle rather than carbon dioxide as prescribed in this contract. From their study a computer program was written that would provide nozzle design criteria for nitrogen as a working gas. Because the basic equations are the same, the USCEC computer program could be used for nozzle design criteria with carbon dioxide as a working gas after modifications are made for carbon dioxide thermodynamic and transport properties. From this program a parametric study to determine the nozzle design criteria has been made. The effects of specific heat ratio on Mach number, boundary layer thickness, displacement thickness, and the radius of the uniform core were determined parametrically. Nozzle design criteria were then determined by plotting the variation of specific heat with temperature onto a family of constant specific heat ratio-Mach number curves. A design curve for Mach number versus nozzle station was found from which all other nozzle characteristics could be defined. Design curves are presented for all nozzles studied. Two phase flow probability is shown for nozzles with Mach numbers greater than 4 at the exit.

Author

A Mach 4 nozzle is recommended in preference to 6 because of the probability of two-phase flow occurring at Mach numbers greater than 4. This nozzle also meets the contract requirements defined for uniform core radius and room temperature plenum chamber conditions. Higher Mach numbers may be possible through plenum chamber heating and boundary layer suction. The effect on the flow field of varying total pressure, total temperature, and wall was considered, and the results presented graphically. Cryopumping requirements were defined for the flow rates anticipated.

TABLE OF CONTENTS

NOMENCLATURE

a	Nozzle radius
K	Degrees Kelvin
M	Mach number
P	Pressure
r	Coordinate
R	Degrees Rankine
T	Temperature
u	Flow velocity
x	Distance along nozzle
γ	Specific heat ratio
δ	Boundary layer thickness
δ^*	Displacement thickness
μ	Viscosity
ρ	Density
ω	Nozzle half-angle

Subscripts:

1	Free stream conditions
0	Stagnation conditions
ec	Expansion core
ref	Reference
w	Wall condition
*	Throat condition, $M = 1$

INTRODUCTION

Developments in the application of cryogenic pumping to low-density wind tunnel use have made possible the duplication of the static pressures encountered at orbital altitudes. The cryopump whose pumping capacity for a particular gas is solely a function of surface area is capable of pumping large volumetric flows resulting from the low operating pressure. To increase its ground-based testing capability, the Marshall Space Flight Center contracted the University of Southern California Engineering Center (USCEC) to present nozzle design criteria for a low-density wind tunnel.

The USCEC developed equations (Reference 1) suitable for determining the nozzle design criteria for a low-density wind tunnel in which nitrogen was the working gas. However, nitrogen requires a gaseous helium (20K) cryopump. While this combination produces a very desirable flow field aerodynamically, a helium refrigerator with the capacity to cryopump large mass flows would be quite expensive. A more economical method is to use a working gas which can be cryopumped at liquid nitrogen temperature (77°K). Carbon dioxide and water vapor can be cryopumped at this temperature. Of the two, carbon dioxide is preferable because higher Mach numbers may be attained with it.

It was the purpose of the research study performed under this contract to (1) account for the differences in the thermodynamic and transport properties between carbon dioxide and nitrogen and (2) design a low-density wind tunnel nozzle using carbon dioxide as a working fluid. Primarily this involved obtaining reliable data for the viscosity, specific heat ratio, and Prandtl number and incorporating these data into the existing USCEC program which iteratively solves the equations. However, because of the interdependence of Mach number, free stream temperature, and specific heat ratio, a parametric solution was necessary.

Having made these changes, nozzle design criteria were obtained for Mach 3, 4, 6, 9, and 12 nozzles. Above Mach 4, two-phase flow is probable when expanding carbon dioxide from near-room temperature. For this reason, design criteria for a Mach 4 nozzle were obtained in addition to the Mach numbers required by the contract. This summary report presents the design criteria, pumping requirements, operating range for nozzles that fit the Marshall Space Flight Center 3-1/2 x 7-foot high-vacuum chamber.

DISCUSSION

USCEC Program

The USCEC computer program written to solve the equations in Reference 1 was available in two forms. In one, the "specified wall" program, the throat radius and nozzle half-angle are input and the isentropic core size and Mach number distribution calculated. In the other, the "specified core" program, an approximate Mach number distribution and the radii of the desired isentropic core at each station are input and the required nozzle coordinates computed. Most of the program modifications have been concentrated on the specified wall program because the nozzle must fit within an existing 3-1/2 x 7-foot vacuum chamber.

Some initial difficulty was experienced because source decks were not available for the subroutines used in the computer program. The listings were received 25 February 1964. Prior to the receipt of the subroutines modifications were being made in the main program, so no serious delay was involved.

Areas of Modification

The modifications to the program were generally those which accounted for the differences in the thermodynamic and transport properties between carbon dioxide and nitrogen. That differences should exist is not surprising because carbon dioxide is a linear triatomic molecule whereas nitrogen is a diatomic molecule. Futhermore, because of the additional vibrational degrees of freedom available to the carbon dioxide molecules, the greatest and most significant variation is in the specific heat ratio. Viscosity differences are also apparent but not as important. Some difference is to be expected in the Prandtl number but again this will have a relatively small effect on the results.

Data for the variation of the specific heat ratio of carbon dioxide with temperature and pressure were obtained from Reference 2. These data are plotted on Figure 1. Specific heat ratio varies with pressure but the variation is slight and may be neglected if the 0.01 atmosphere data are selected. No data were obtained below 200°K.

Viscosity data were obtained for temperatures to 175°K from References 2 and 3. Figure 2 shows the comparison among the experimental data and the empirical Sutherland and linear laws. Both empirical forms give very good fits, so there is no advantage to using the more complex Sutherland equation.

Figure 3 shows the variation of Prandtl number with temperature. These data obtained from Reference 1 indicate a change in Prandtl number occurs with temperature, increasing in slope with decreasing temperature.

Limit of Single-Phase Flow

Expanding a gas from rest to a high Mach number produces a severe decrease in temperature. This temperature drop becomes significant if a phase line for the particular gas is approached. In general the position of the phase line is a function of both temperature and pressure, the decreasing pressure shifting the phase transition to a lower temperature. Figure 4, taken from Reference 4, shows the phase diagram for carbon dioxide superimposed on an isentropic expansion plot, assuming an initial total temperature of 228°K or 518°R. It is evident from this diagram that Mach numbers greater than 4 using room temperature carbon dioxide will be impossible to achieve. Increasing the total temperature will increase the free stream temperature and allow higher Mach numbers to be reached. But an upper limit to this solution exists. At about 1000°C carbon dioxide begins to decompose into carbon monoxide and oxygen, neither of which can be cryopumped at liquid nitrogen temperature.

A further effect occurs as the total temperature is increased. Figure 5 shows the radius of the isentropic core as a function of nozzle length for a Mach 4 nozzle having total temperatures of 540°R, 600°R, 750°R, and 1000°R. Large increases in boundary layer thickness are evident with the increase in total temperature until at 1000°R the boundary layer completely fills the nozzle. From this fact it must be concluded that if the total temperature is to be increased, boundary layer removal by suction is necessary. For these reasons and with due regard for the requirement that "near room temperature carbon dioxide" be used, it is strongly recommended that a Mach 4 rather than a Mach 6 nozzle be considered for the Marshall Space Flight Center low-density wind tunnel.

Temperature Variation of Specific Heat Ratio

The principal difficulty encountered in using the USCEC program for obtaining nozzle design criteria for system using carbon dioxide as a working gas centered around the variable specific heat ratio of carbon dioxide. Because the specific heat ratio of nitrogen remains relatively constant over the temperature range considered the program was written to treat the specific heat ratio as an input constant. This leads to considerable error in computing the Mach number for carbon dioxide if the variation in specific heat ratio is ignored.

The problem is apparent if the equation from which Mach number is calculated is examined.

$$M^2 = \frac{\gamma+1}{\gamma-1}\left[(M^2)\ (A/A_*)^2\right]^{\frac{\gamma-1}{\gamma+1}} - \frac{2}{\gamma-1} \qquad (1)$$

This equation implicitly assumes that γ at any station is a constant up to that point. Because γ is a function of temperature and because of the temperature drop resulting from the expansion in the nozzle, the equation becomes less and less accurate as the distance from the throat is increased.

Analytic Solution

Two solutions to this problem were considered. One, an analytic solution, makes use of the following equation:

$$M_n^2 = \left[\left(\frac{M_n}{M_{n-1}}\right)^2 \left(\frac{A_n}{A_{n-1}}\right)^2\right]^{\frac{\bar{\gamma}-1}{\bar{\gamma}+1}} \left[\left(\frac{2}{\bar{\gamma}-1} + M_n^2 - 1\right)\right] - \frac{2}{\bar{\gamma}-1} \tag{2}$$

where the subscripts refer to particular nozzle stations and $\bar{\gamma}$ is an average between stations n and n-1. While not exact the use of an average γ for the volume bounded by A_n and A_{n-1} rather than a constant γ over the entire length should provide increased accuracy. An effort was made to program and include this approach during the contract period. However, this could not be done and the parametric study described in the following paragraphs was employed to determine the nozzle design criteria.

Parametric Method

Simultaneously with the analytic approach a parametric study was initiated. To facilitate this method a linear fit was made of the specific heat ratio data on Figure 1.

$$\gamma = aT + b$$
$$a = 0.000352 \qquad b = 1.473 \tag{3}$$

Then with the equation for temperature as a function of specific heat ratio and Mach number

$$T = \frac{T_o}{1 + \frac{\gamma-1}{2} M^2} \tag{4}$$

a simultaneous solution for temperature was made giving

$$T = \frac{-\left(1 + \frac{b-1}{2} M^2\right) + \sqrt{\left(1 + \frac{b-1}{2} M^2\right)^2 + 2aM^2T_o}}{aM^2} \tag{5}$$

taking the positive square root. The solutions of equation 5 from Mach 1 to 13 are plotted on Figure 6.

Then, with Equation 3, the γ for carbon dioxide at the plenum chamber temperature was computed. Next the temperature at the chosen exit Mach number was calculated from Equation 5 and the γ associated with this temperature from Equation 3. (At this point it should be made clear that exit temperatures were in some cases far below the temperatures for which specific heat ratio data existed for carbon dioxide and thus the γ obtained was the result of a linear extrapolation of the available data.) Then, with the bounding specific heat ratios, computer runs were made using constant specific heat ratios between and including the boundaries.

From these data a plot was made of Mach number versus nozzle length for each of the γ's. Now having γ as a function of temperature (Equation 3) and temperature as a function of Mach number (Equation 5), it was possible to plot onto the constant γ-Mach number curves, a single curve accounting for the variation of specific heat ratio with temperature. This curve is called the design curve in subsequent plots and its intersection with the required Mach number the design point. This procedure was followed for all Mach numbers.

Core-Wall Comparison

Because of its direct applicability to the present problem, the specified wall program was used exclusively. At one point a comparison between the specified wall and specified core programs was made. The results indicated that large discrepancies existed between the two programs. Whether this was the result of inherent inaccuracies in computation method or some other problem was not determined since the specified core programs was not to be used in subsequent nozzle design criteria computations.

Hand Computation

To verify the computer results, a check case was initiated on the specified wall program. The initial conditions were computed by setting the plenum chamber pressure at 0.01 psi and the temperature at 621°R. From these values the mean molecular velocity and density were then computed. The gas flow tables (Reference 4) with $\gamma = 1.30$ were then used to find the same quantities for a Mach number of 1. It was decided that three stations down the nozzle separated by .333 ft would be checked.

The ratio of cross-sectional areas at the station and the throat could be used as a first guess in the hand calculation. To reduce the time required to complete the iterations, the Mach number computed by the program was used as the initial input. With this Mach number and using gas flow tables, the temperature, pressure, density and velocity could all be computed for the particular station and these values then used to compute a Mach number. Since the program was iterative, the computed Mach number was compared with the initial Mach number for that station. If the difference was within acceptable limits, the computed value was accepted and the process repeated at the next station. If the computed value was not acceptable, it was used as the new input. This procedure was repeated until the results were acceptable. Convergence of the iteration was rapid the Mach number usually being within a few hundreds of the prior iteration after three trials. The results along with the initial conditions are presented on the following page.

Nozzle Station	Mach Number	
	Hand Computed	Machine Computed
X_1 = 0.333 ft	1.95	1.987
X_2 = 0.666 ft	2.35	2.409
X_3 = 0.999 ft	2.69	2.725

Initial Conditions:

T_w	P_{t_o}	T_{t_o}	T_R	ω	μ_{ref}
360^oR	0.01 psi	621^oR	540^oR	13^o	3.17×10^{-7} $\frac{slugs}{ft\text{-}sec}$

A^*	μ_H	Pr	ΔX
0.212	2.13×10^{-7} $\frac{slugs}{ft\text{-}sec}$	0.86	0.333 ft

Nozzle Design Criteria

By using the parametric approach described earlier in the report, nozzle design criteria were developed for nozzles having 3, 4, 6, 9, and 12 exit Mach numbers. These data are presented on Figures 7-36. For each Mach number, the effect of γ on Mach number, displacement thickness, boundary layer thickness and the radius of the isentropic core are presented. In addition, the proposed nozzle coordinates are shown with the isentropic core and Mach number distribution obtained from the parametric study.

With regard to the plots of boundary layer thickness and the radius of the isentropic core, it should be noted that the data from which these plots were obtained are the result of discrete calculations rather than from a continuous equation. For this reason smooth curves were not obtained and symbols were used rather than fairing smooth curves through the points.

All nozzles were designed for a total temperature of 540^oR. Boundary layer control was accomplished by cooling the nozzle walls to 360^oR. This is slightly above the freezing point of carbon dioxide at atmospheric pressure and with the reduced pressure in the nozzle there is no danger of cryopumping the flow onto the nozzle walls. Boundary layer control by means of suction was not considered since adequate core sizes could be obtained without it.

Mach 3

The data for the Mach 3 nozzle are presented on Figure 7 through 12. The total pressure is 1.9342×10^{-3} psi or 100 microns. The throat radius is 0.7 feet and the nozzle length obtained from the design curve (Figure 6) is 3.89 feet. The nozzle half-angle is 13°. The boundary layer thickness (Figure 8) is 0.455 feet giving a uniform core of 1.15 feet at the exit. The static pressure at the exit is plotted on Figure 11 for the highest and lowest γ and is of the order of 5.5×10^{-5} psi. The nozzle coordinates proposed for a Mach 3 nozzle are presented on Figure 12 showing the radius of the uniform core and the Mach number distribution along the nozzle corrected for the variation or γ .

Mach 4

Figures 13 through 18 contain the data obtained for a Mach 4 nozzle. For a total pressure of 1.9342×10^{-3} psi a Mach number of 4 was reached at a nozzle length of 4.90 feet (Figure 13). The throat radius is 0.319 feet and the nozzle half-angle 12°. The boundary layer thickness is 1.1 feet (Figure 14) giving a uniform core of 0.595 feet (Figure 16). The exit static pressure is about 1.2×10^{-5} psi (Figure 17). Design coordinates for a Mach 4 nozzle are given on Figure 18 with the uniform core radius and Mach number distribution corrected for the variation of γ with temperature.

Mach 6, 9, and 12

The data for Mach 6, 9, and 12 nozzles are presented on Figures 19-36. Reference to Figures 24, 30, and 36 will show that for all three nozzles two-phase flow will result at about Mach 4 even allowing for supercooled flow. Results at Mach 6, 9, and 12 are therefore of academic interest only and do not relate to nozzles which could be physically constructed.

Effect of Varying P_{TO}, T_{TO}, and T_W

To determine the effect on the Mach 4 nozzle design criteria results, the total pressure, total temperature, and the wall temperature were varied. Large variations were taken to accentuate the effects. Figure 37 shows the change in Mach number resulting from total temperatures of 540°R, 600°R, and 750°R. Only a very small change can be observed. Referring to Figure 5, it appears that for the same total temperatures there is a large decrease in core size. Thus large variation in the total temperature can be tolerated if core size can be sacrificed.

Figures 38 and 39 point out the effect of a change in total pressure on the Mach number and the uniform core radius. As expected increasing P_{TO} from 2×10^{-3} psi to 2×10^{-2} psi causes a significant increase in core size and a resulting increase in Mach number as the displacement thickness is reduced. However, the increase in the static pressure reduces the usefulness of flow field as a low-density stream. Figure 40 shows the decrease in Mach number resulting from an increase in wall temperature.

This is not severe and so small variations in wall temperature will not be significant during the operation of the nozzle.

Cryopumping Requirements

The maximum weight flow anticipated is 0.02 lb/sec or about 9 grams/sec. An estimate of the cryopumping capacity based on a cryopanel area of 86 square feet or 82,500 cm^2 is 500,000 liters/sec. The pumping speed of a liquid nitrogen-cooled surface was taken to be 6 liters/sec-cm^2. A cryopanel covering half of the chamber was used in the calculation and only the interior surface considered. Thus, the cryopumping capacity of the system appears to be more than adequate for the condensable flow.

Non-condensable impurities in the carbon dioxide must be pumped by the diffusion pumps. An excessive amount of non-condensables could cause an increase in static pressure sufficient to disrupt a run. For this reason it is recommended that liquid rather than bottled carbon dioxide be used. The impurity content of liquid carbon dioxide is of the order of parts per billion rather than the usual parts per million impurity content of bottled gaseous carbon dioxide. Trace non-condensables of less than 1 ppm can easily be pumped by the available diffusion pumps.

COMMENTS AND RECOMMENDATIONS

The design criteria for the Mach 4 nozzle were obtained by modifying the USCEC program to account for the different thermodynamic and transport properties of carbon dioxide. Few data were available for carbon dioxide below 360°R. For this reason the accuracy of the results can only be as accurate as the extrapolated data. That the results should be dependent on such scanty data is unfortunate but unavoidable. However, an examination of the parametric curves will indicate that even if the specific heat ratio is in error by 10 per cent, a small change in Mach number will result.

The success of the USCEC in comparing the theoretical and experimental data resulting from their study is evidence of the validity of the method employed to obtain the nozzle design criteria. Having no experimental data with which to compare the results of this study, it is not possible to give any indication of its accuracy. But there is no reason why there should be any greater variance in the results of this study than in those of USCEC.

While it was pointed out in the text that Mach numbers greater than 4 cannot be achieved with room temperature carbon dioxide, increasing plenum chamber temperature and applying boundary layer control through suction to obtain higher Mach number is a definite possibility. It is recommended that this possibility be investigated in detail as higher Mach numbers are desirable.

Aside from the many significant force, pressure, and heat transfer studies which can be conducted in a low-density facility, the study of the flow properties of fluid other than air or nitrogen should be of great interest. Particularly since the operation of vehicles in other than terrestrial atmospheres is imminent. It is certain then that such a facility would be able to contribute significantly to the mission of the Marshall Space Flight Center and the National Aeronautics & Space Administration.

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VARIATION OF SPECIFIC HEAT RATIO WITH TEMPERATURE AND PRESSURE

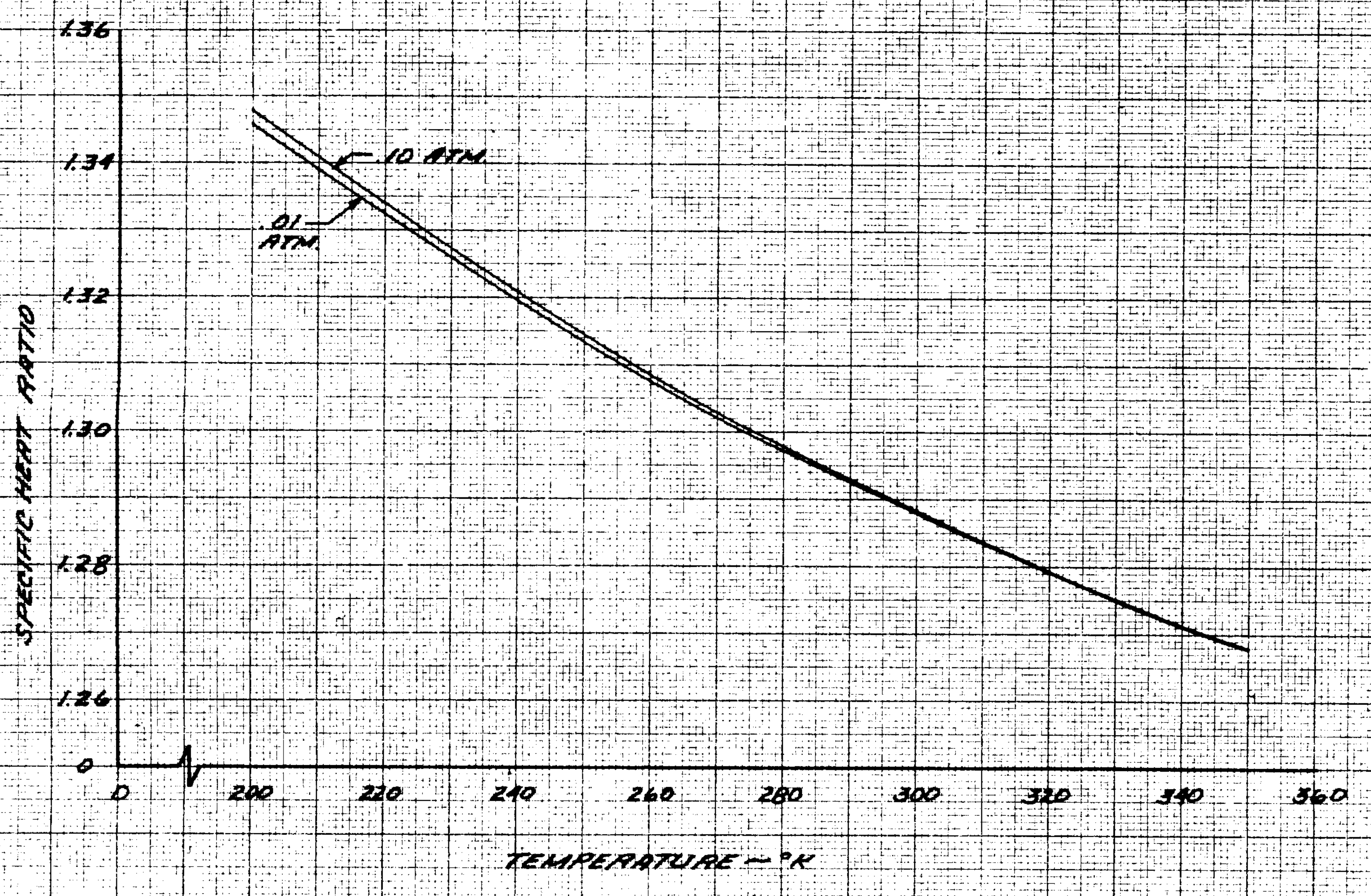


FIGURE 1

CKD BY TMC

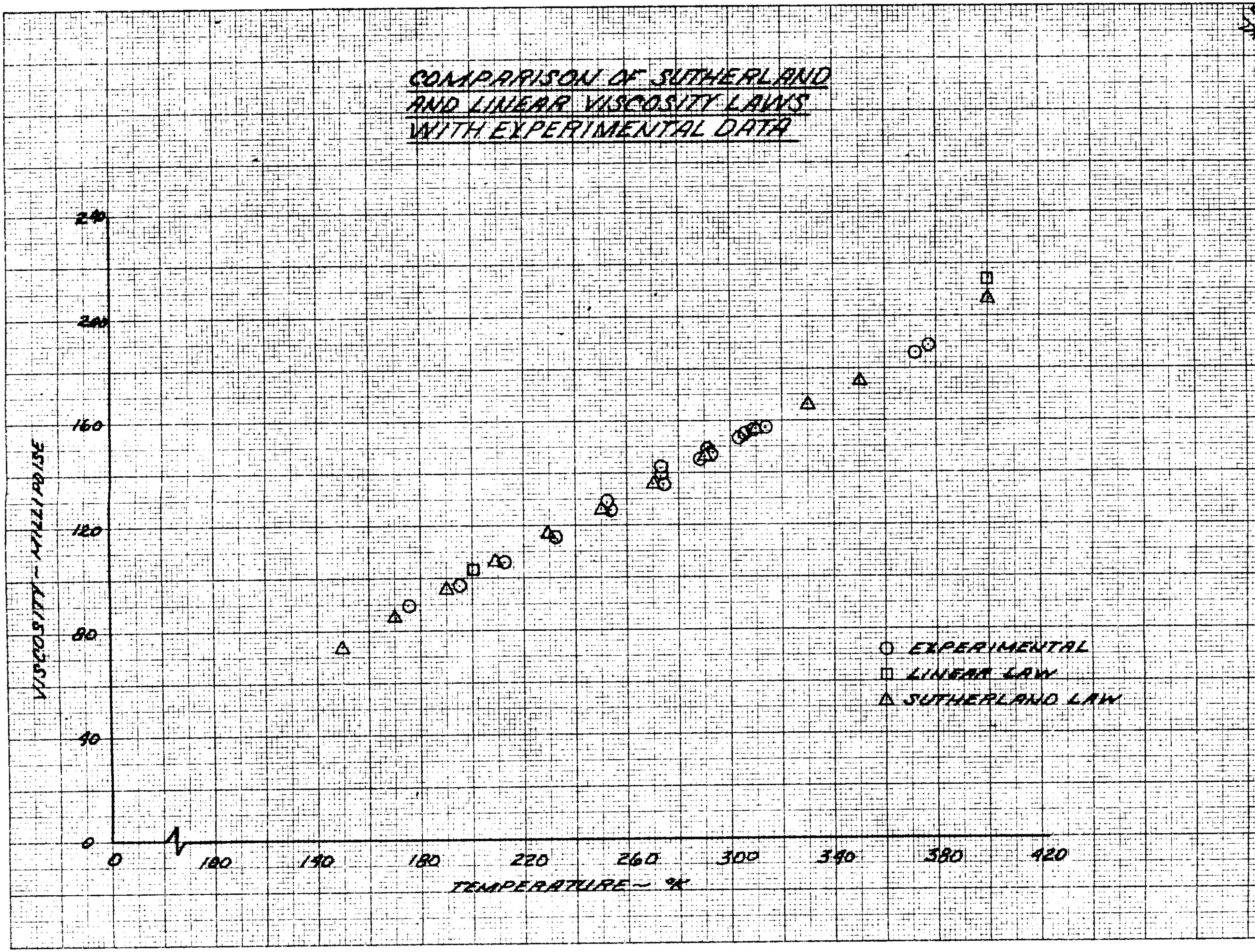

COMPARISON OF SUTHERLAND
AND LINEAR VISCOSITY LAWS
WITH EXPERIMENTAL DATA
VISCOSITY – MILLIPOISE
240
200
160
120
80
40
0
0
100
140
180
220
260
300
340
380
420
TEMPERATURE ~ °K
EXPERIMENTAL
LINEAR LAW
SUTHERLAND LAW

FIGURE 2

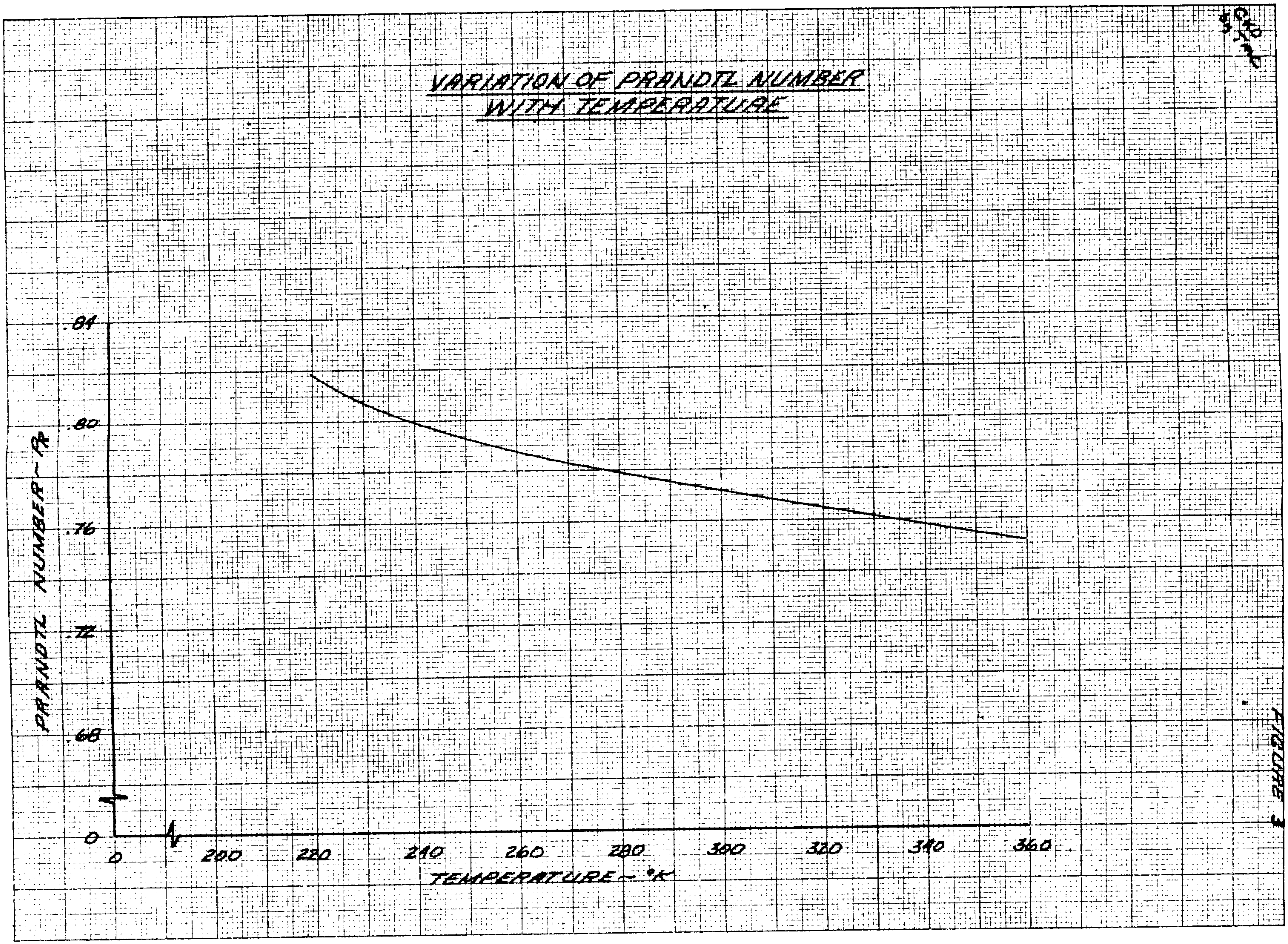
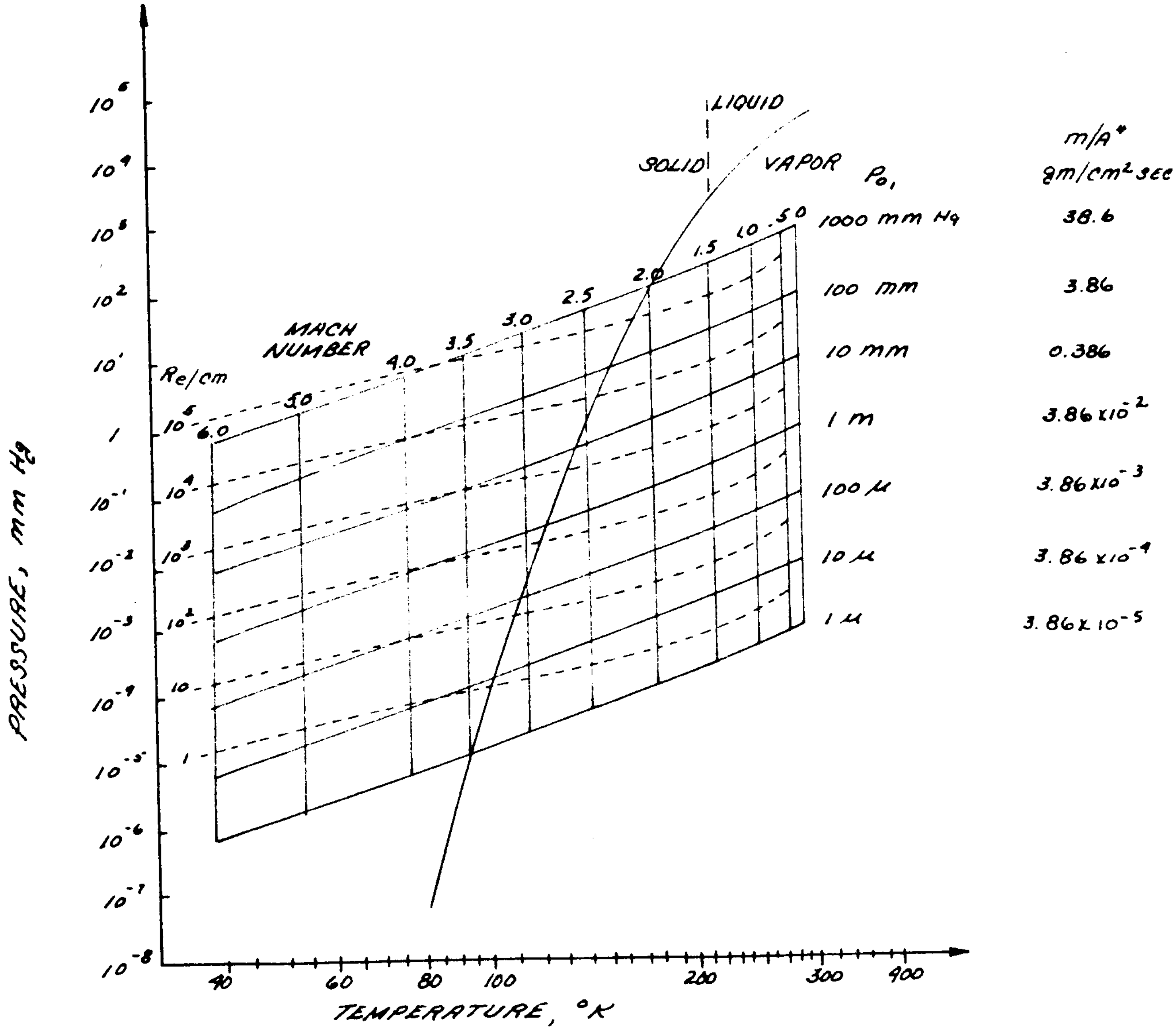


FIGURE 3
VARIATION OF PRANDTL NUMBER
WITH TEMPERATURE
PRANDTL NUMBER ~ Pr
.84
.80
.76
.72
.68
0
0
200
220
240
260
280
300
320
340
360
TEMPERATURE ~ °K
CKD

PRESSURE VERSUS TEMPERATURE FOR ISENTROPIC EXPANSION OF CO_2 FROM $T_c = 288°K$



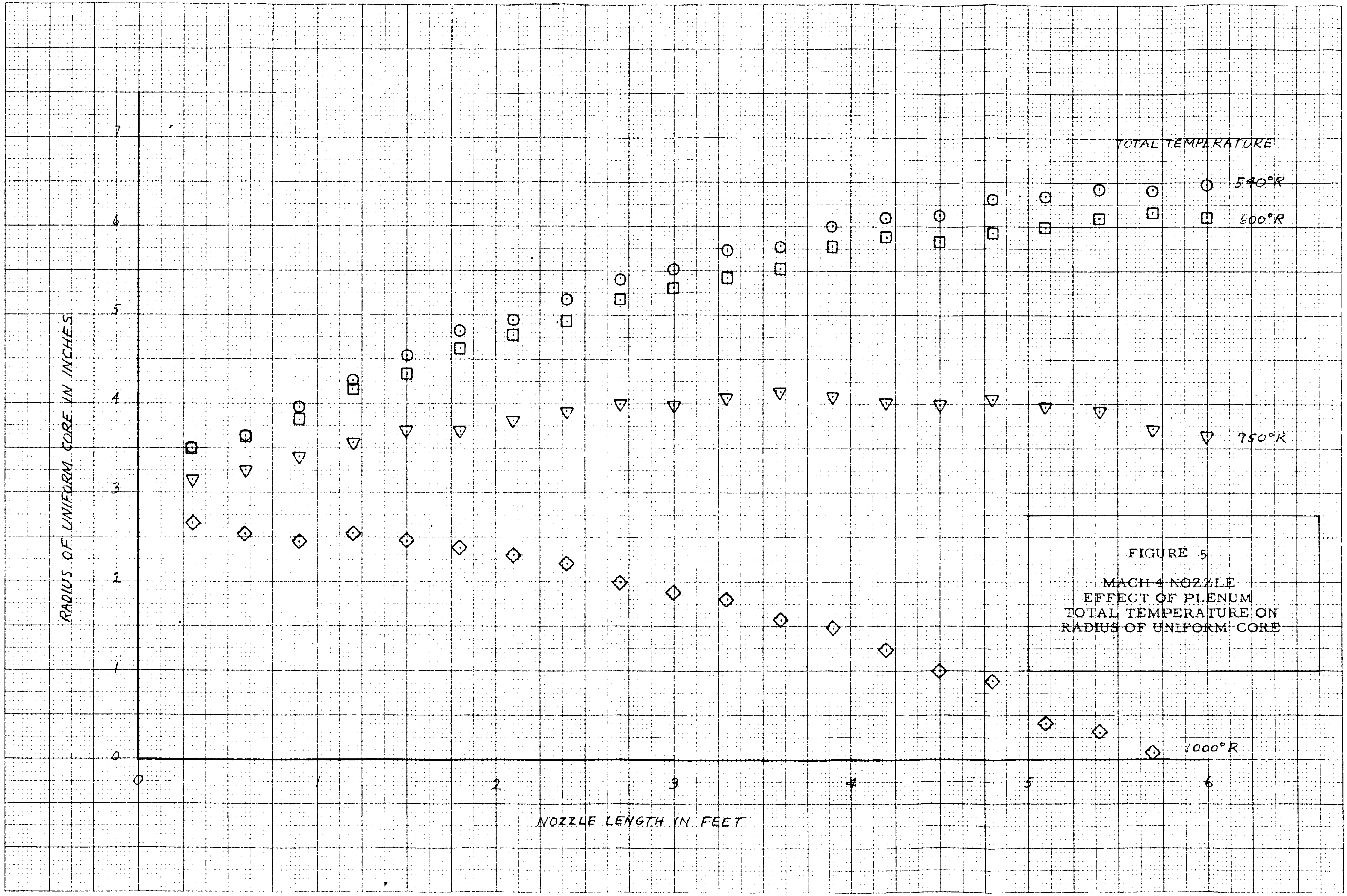


FIGURE 5

MACH 4 NOZZLE
EFFECT OF PLENUM
TOTAL TEMPERATURE ON
RADIUS OF UNIFORM CORE

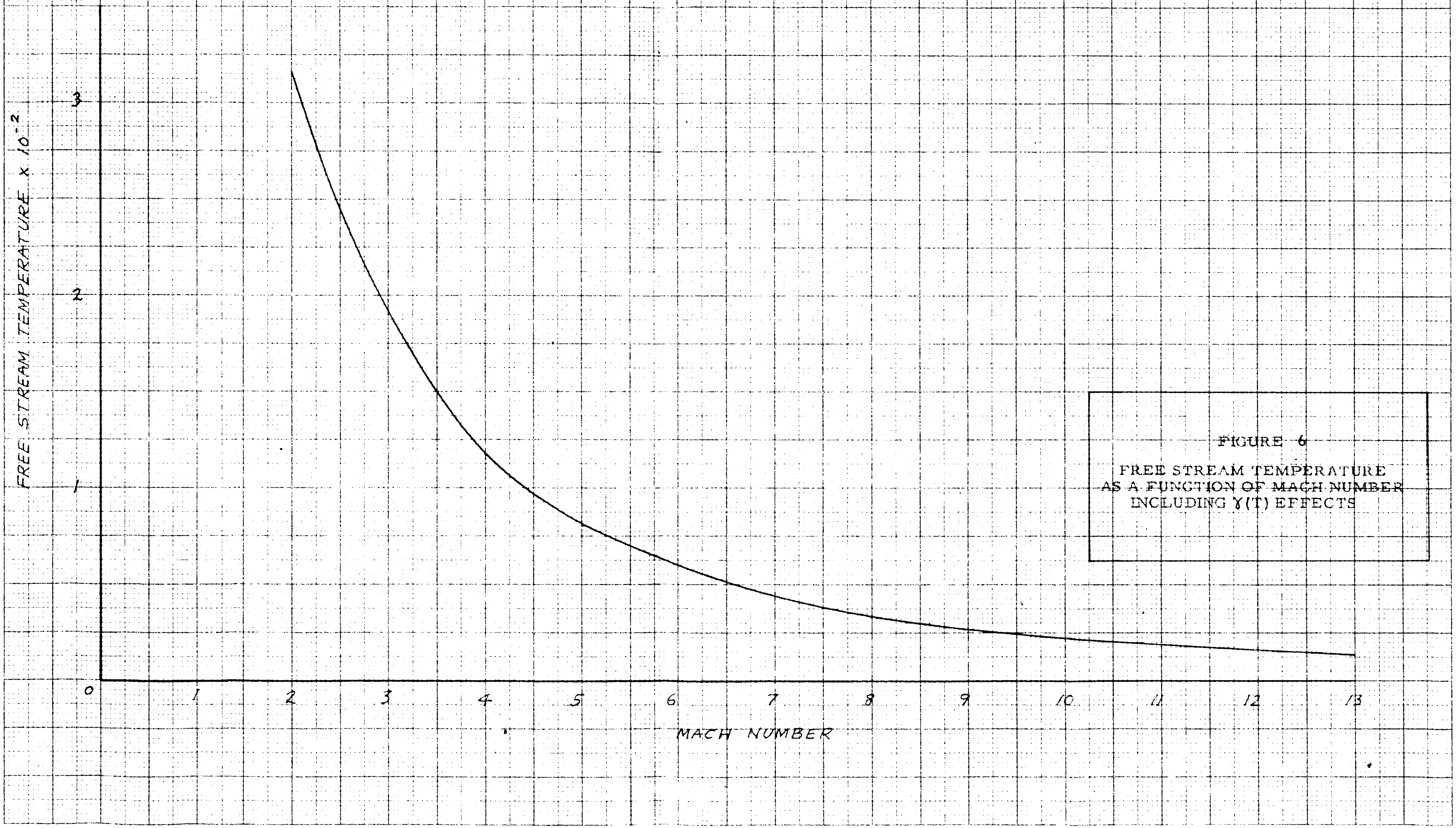


FIGURE 6

FREE STREAM TEMPERATURE
AS A FUNCTION OF MACH NUMBER
INCLUDING $\gamma(T)$ EFFECTS

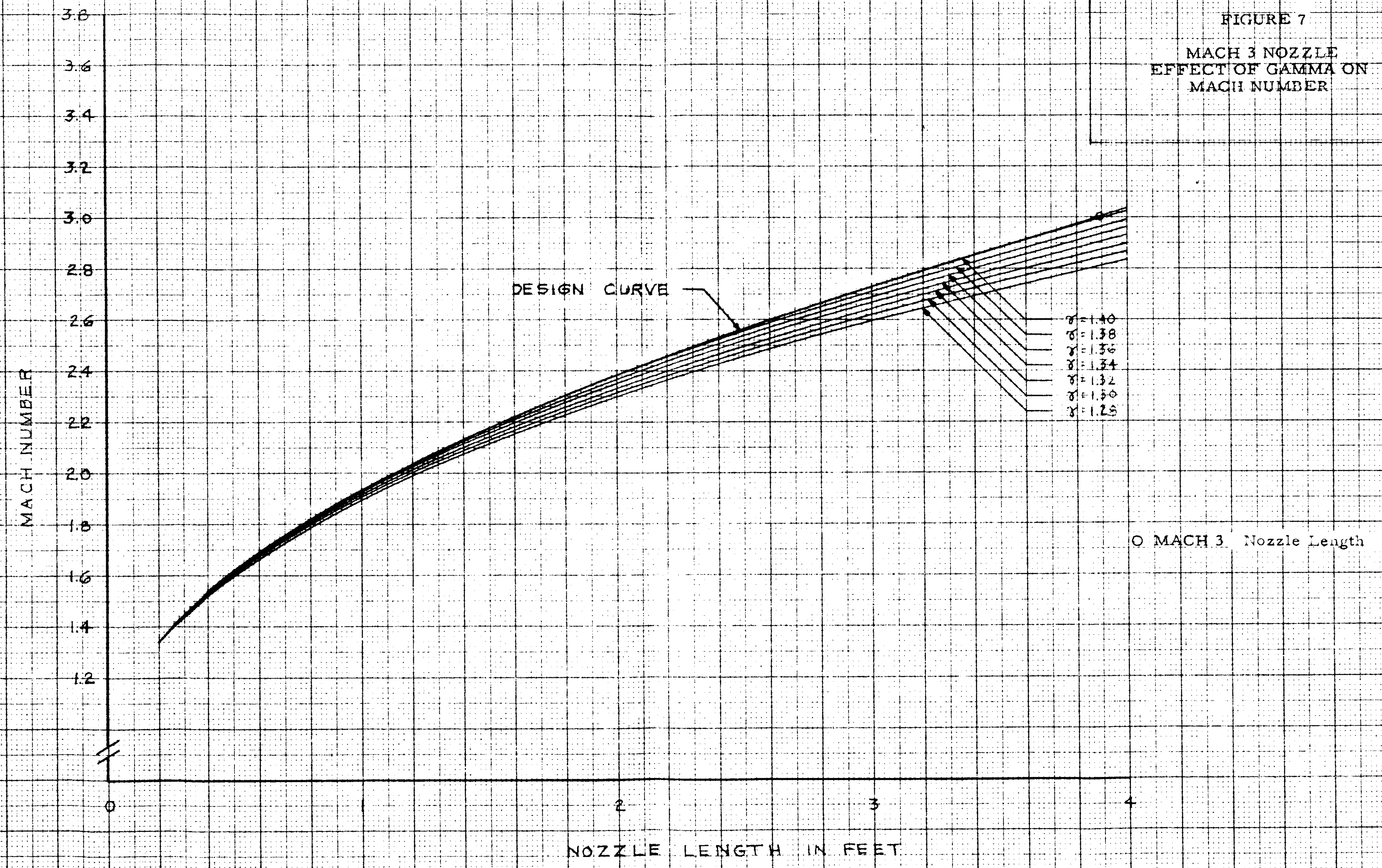


FIGURE 7

MACH 3 NOZZLE
EFFECT OF GAMMA ON
MACH NUMBER

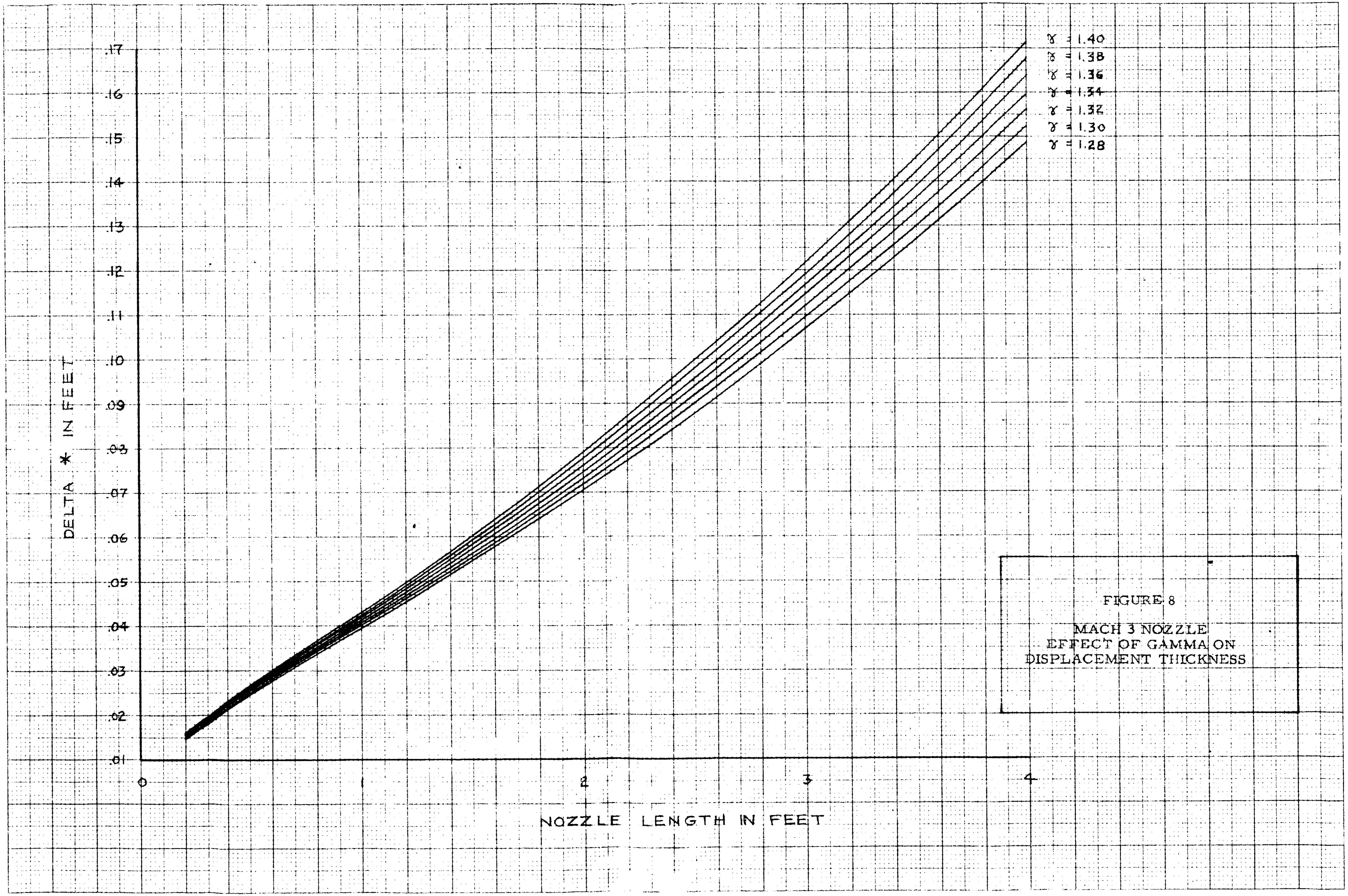


FIGURE 8

MACH 3 NOZZLE
EFFECT OF GAMMA ON
DISPLACEMENT THICKNESS

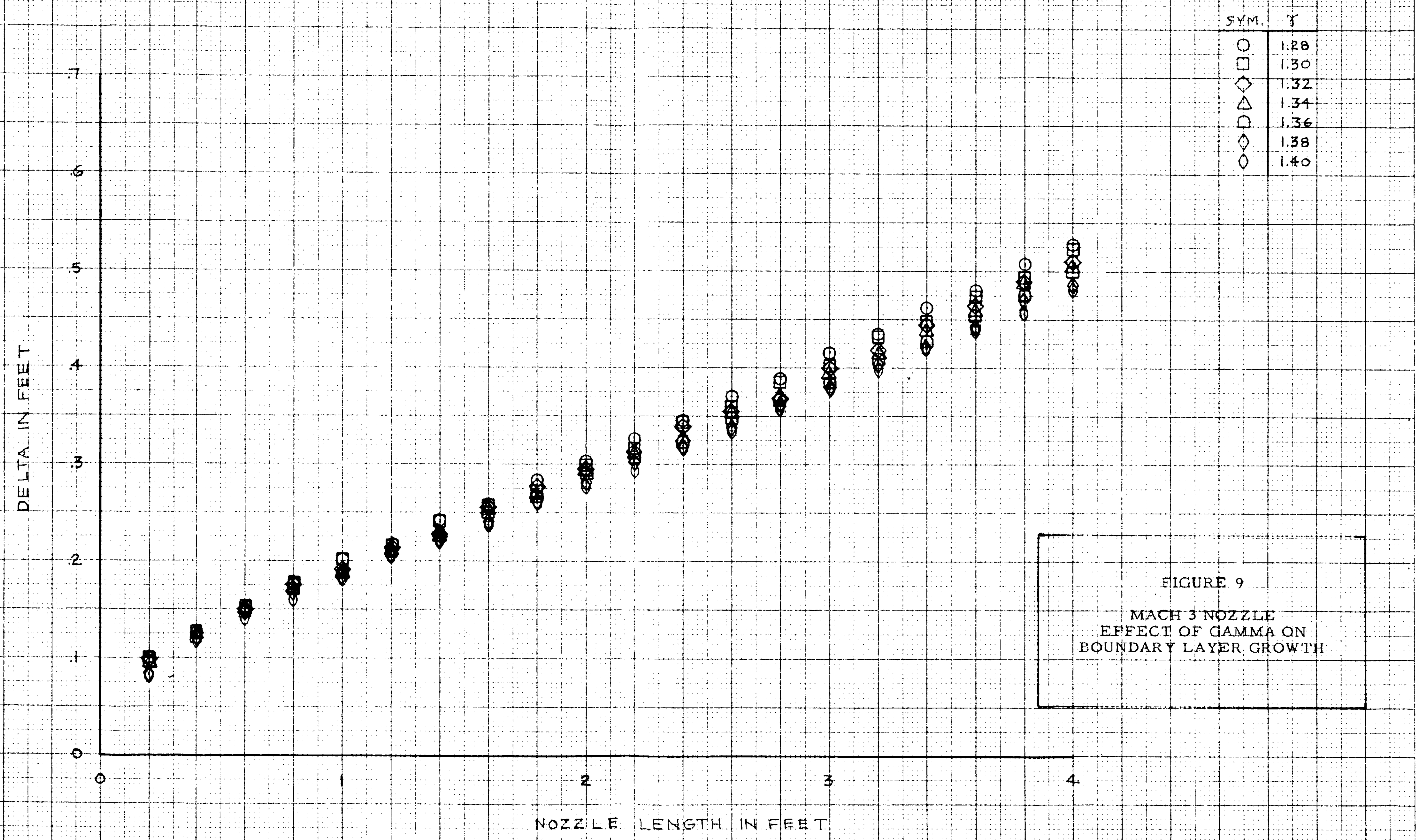


FIGURE 9

MACH 3 NOZZLE
EFFECT OF GAMMA ON
BOUNDARY LAYER GROWTH

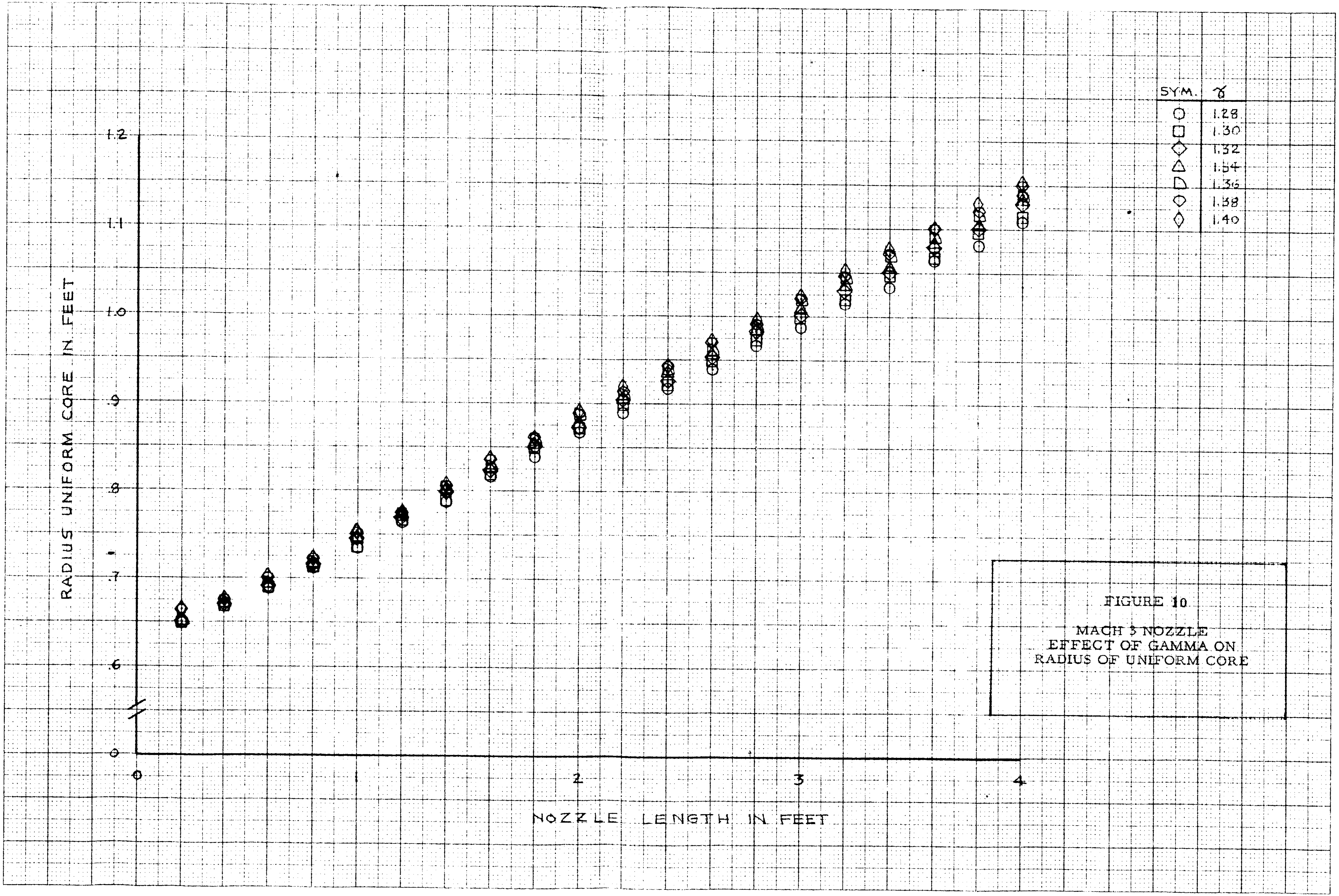


FIGURE 10

MACH 3 NOZZLE
EFFECT OF GAMMA ON
RADIUS OF UNIFORM CORE

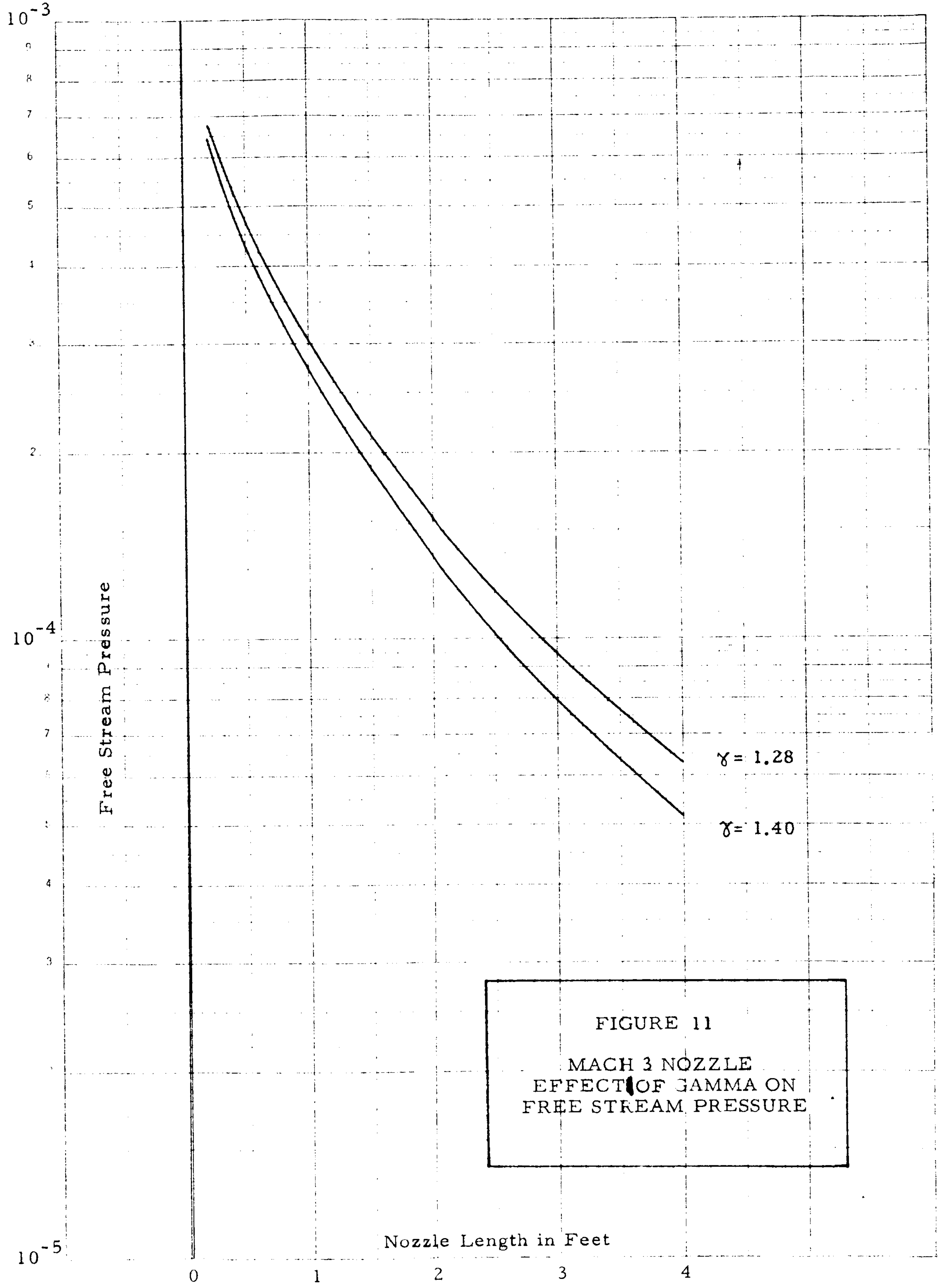


FIGURE 11

MACH 3 NOZZLE
EFFECT OF GAMMA ON
FREE STREAM PRESSURE

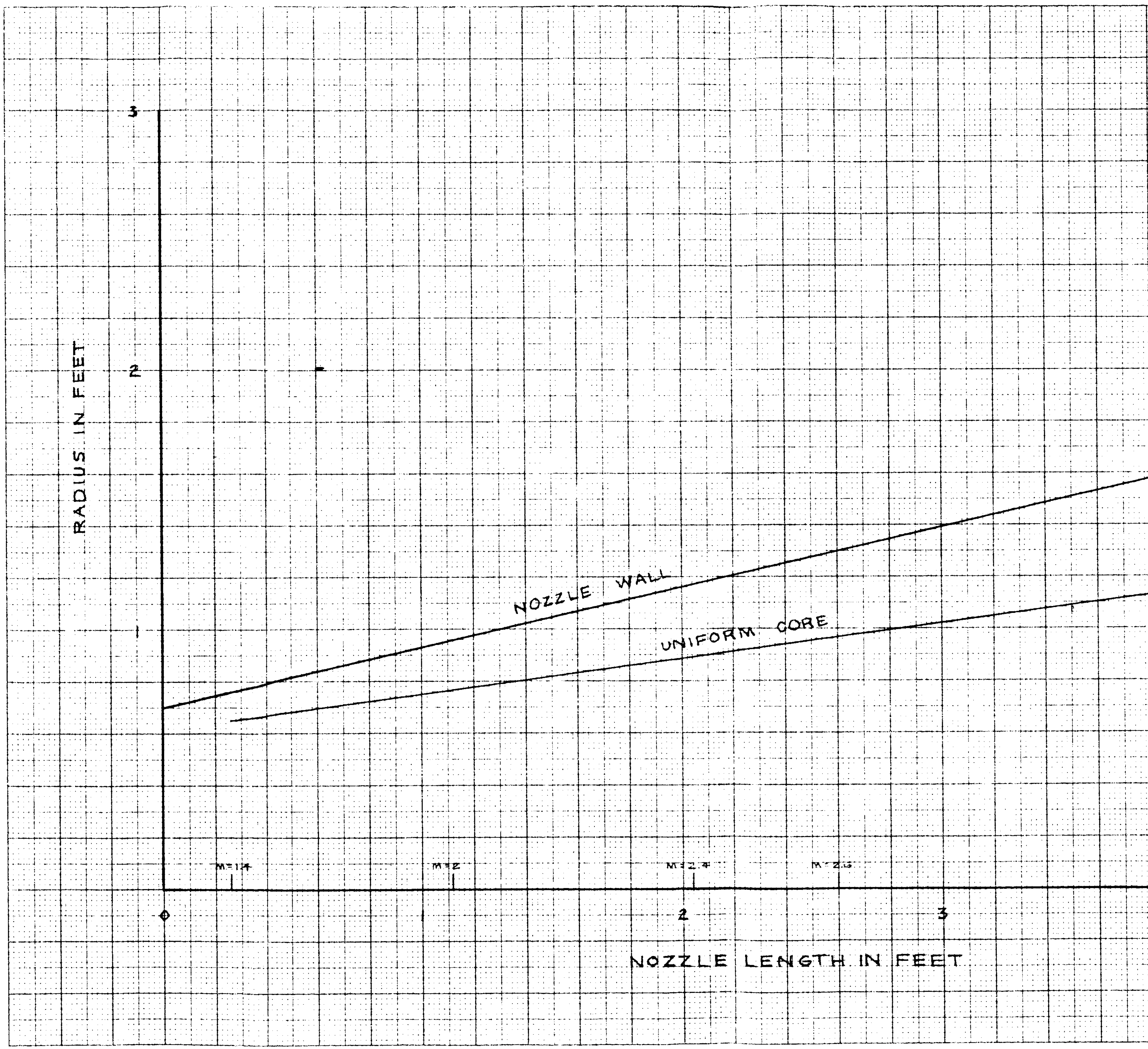


FIGURE 12

MACH 3 NOZZLE COORDINATES AND RADIUS OF UNIFORM CORE

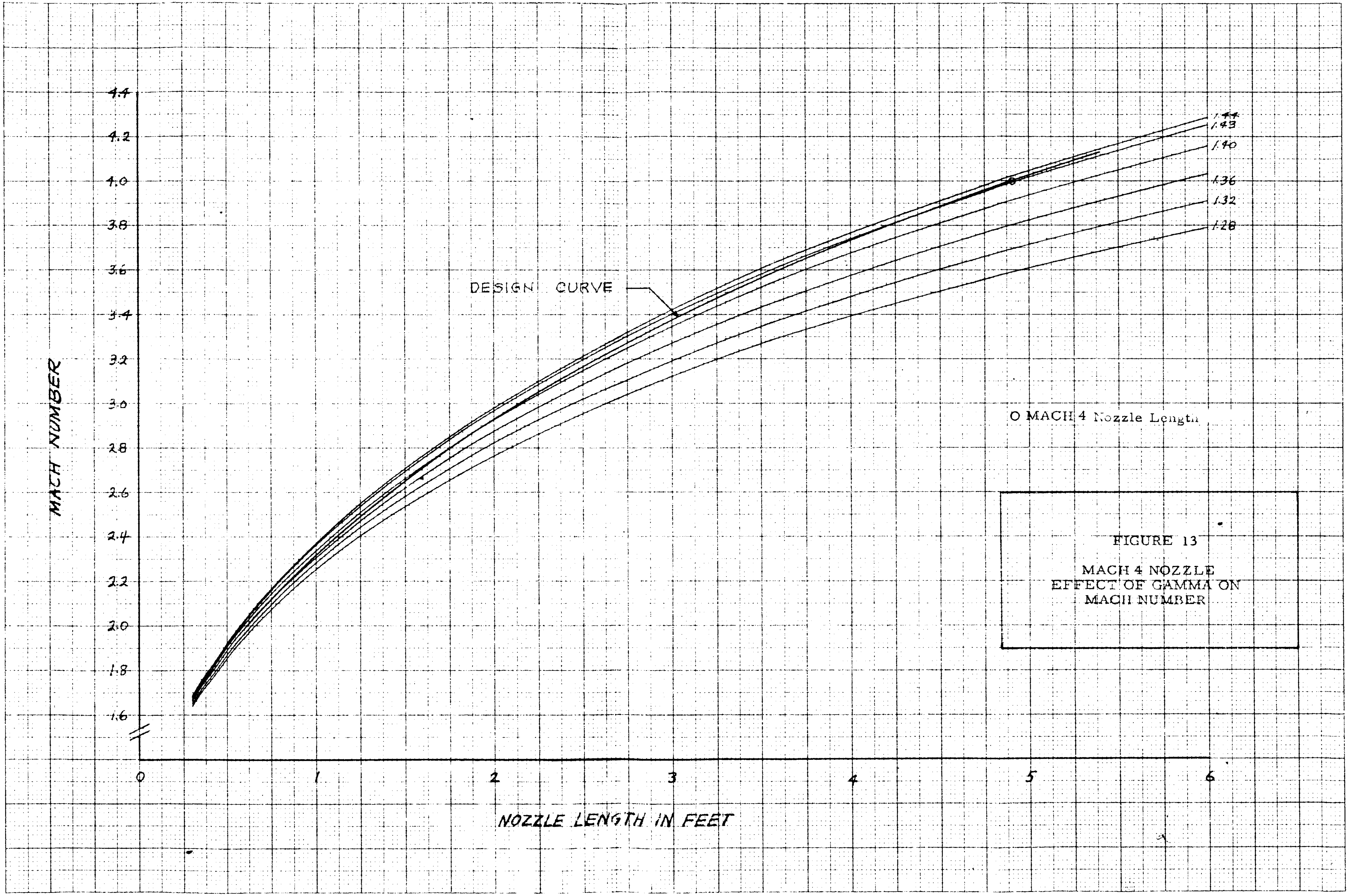


FIGURE 13

MACH 4 NOZZLE
EFFECT OF GAMMA ON
MACH NUMBER

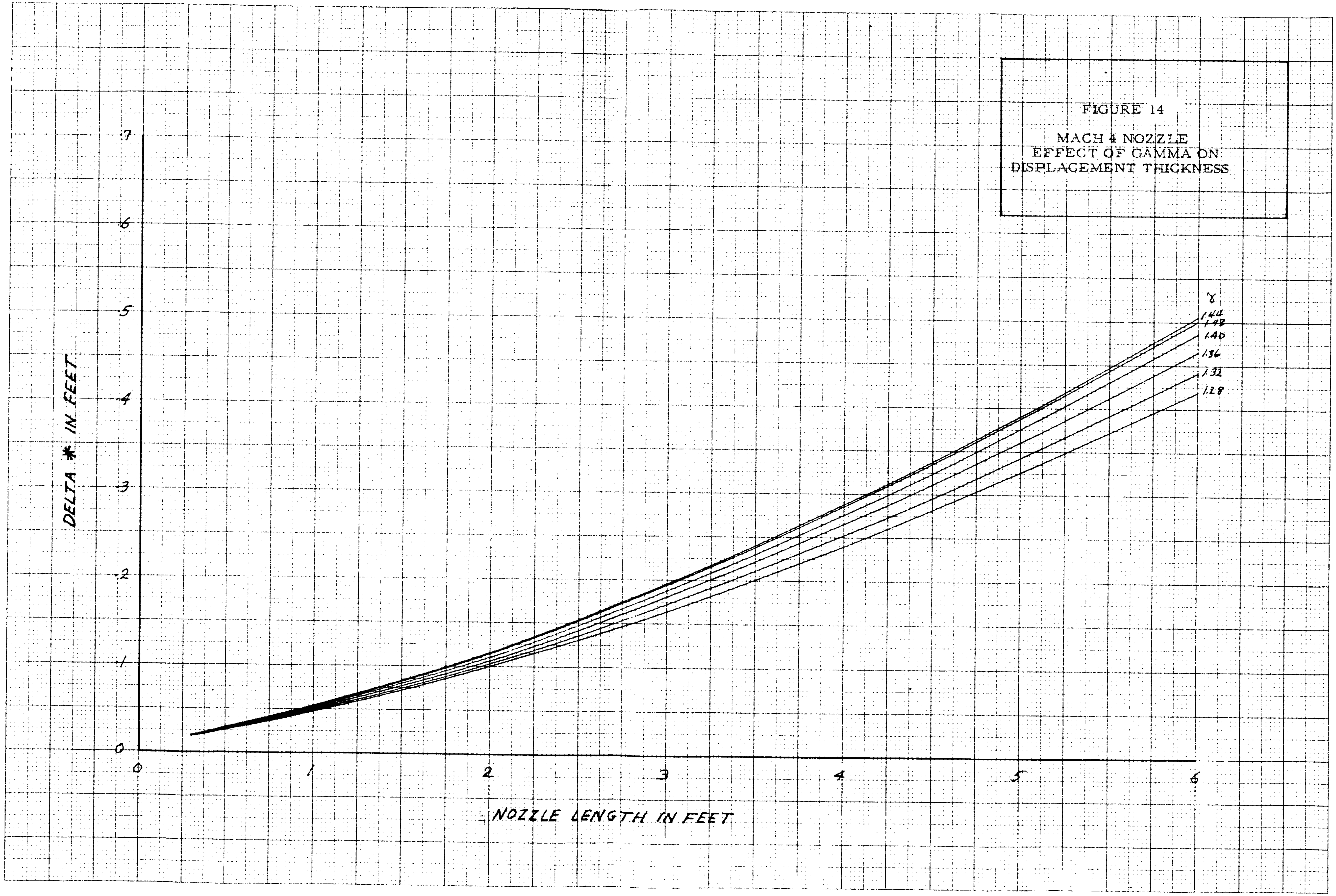


FIGURE 14
MACH 4 NOZZLE
EFFECT OF GAMMA ON
DISPLACEMENT THICKNESS
γ
1.44
1.48
1.40
1.36
1.32
1.28
DELTA * IN FEET
.7
.6
.5
.4
.3
.2
.1
0
0
1
2
3
4
5
6
NOZZLE LENGTH IN FEET

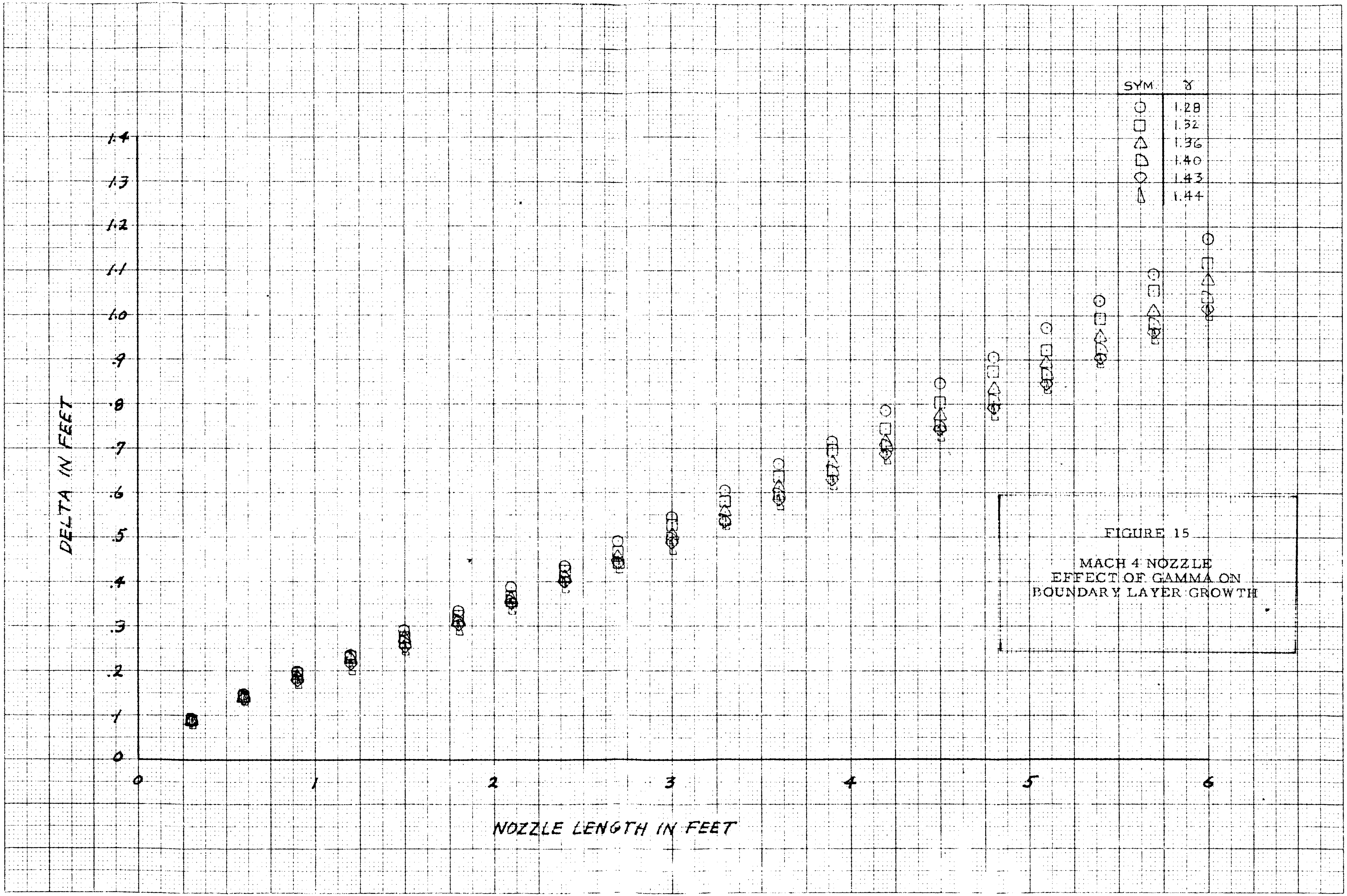


FIGURE 15

MACH 4 NOZZLE
EFFECT OF GAMMA ON
BOUNDARY LAYER GROWTH

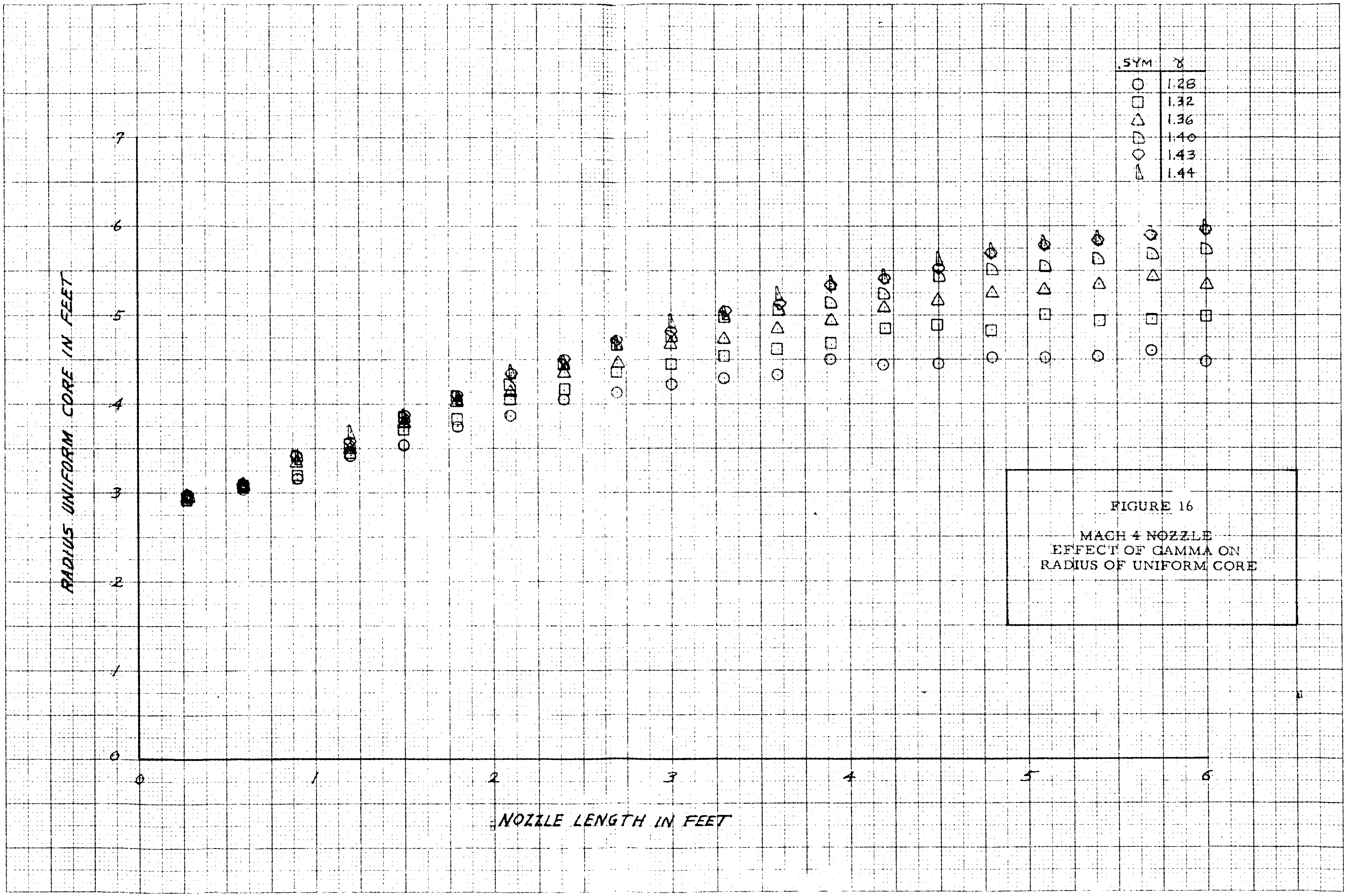


FIGURE 16

MACH 4 NOZZLE
EFFECT OF GAMMA ON
RADIUS OF UNIFORM CORE

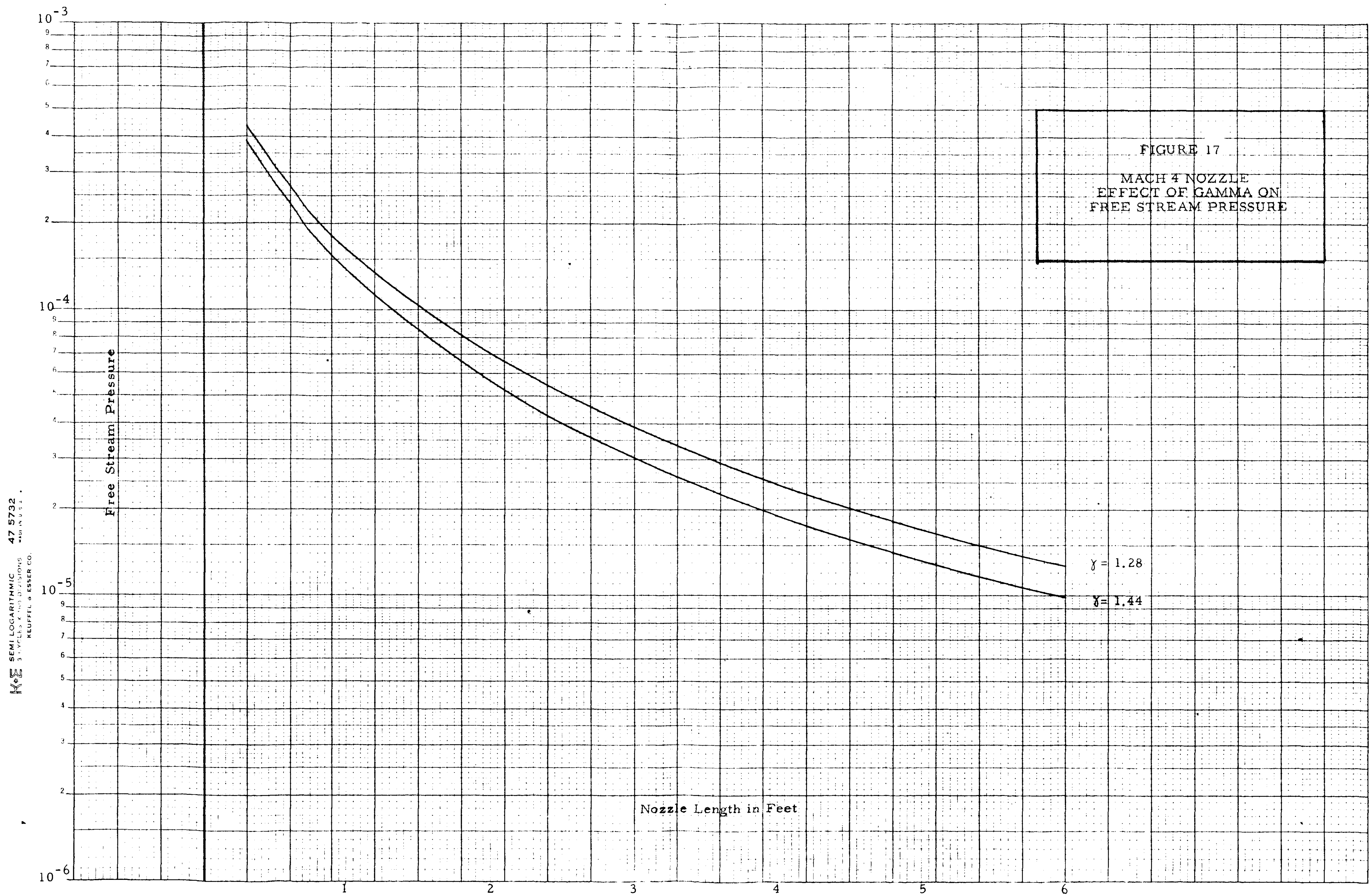


FIGURE 17
MACH 4 NOZZLE
EFFECT OF GAMMA ON
FREE STREAM PRESSURE
$\gamma = 1.28$
$\gamma = 1.44$
Free Stream Pressure
Nozzle Length in Feet
10^{-3}
10^{-4}
10^{-5}
10^{-6}
1
2
3
4
5
6
K&E SEMI-LOGARITHMIC 47 5732
KEUFFEL & ESSER CO.

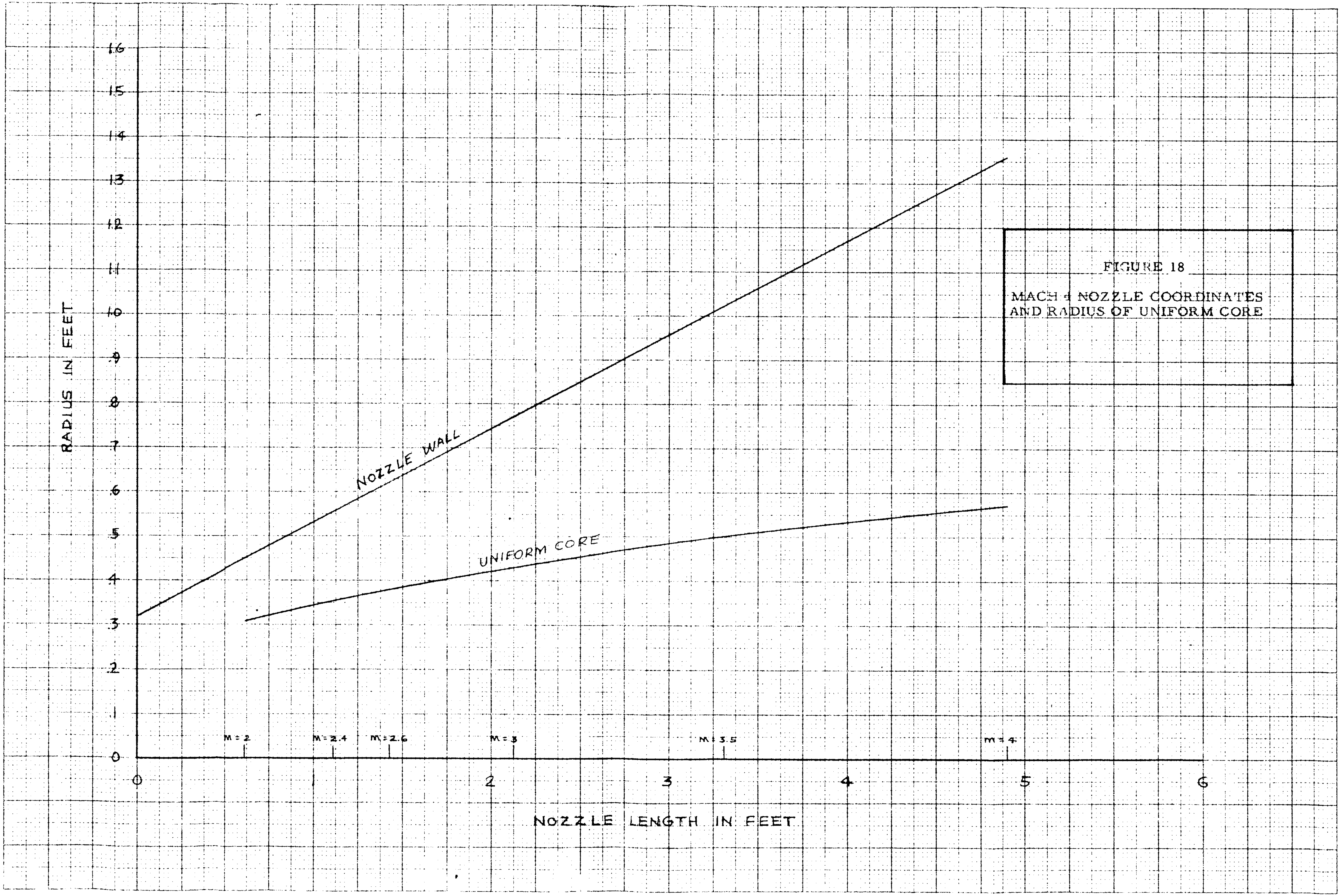


FIGURE 18

MACH 4 NOZZLE COORDINATES AND RADIUS OF UNIFORM CORE

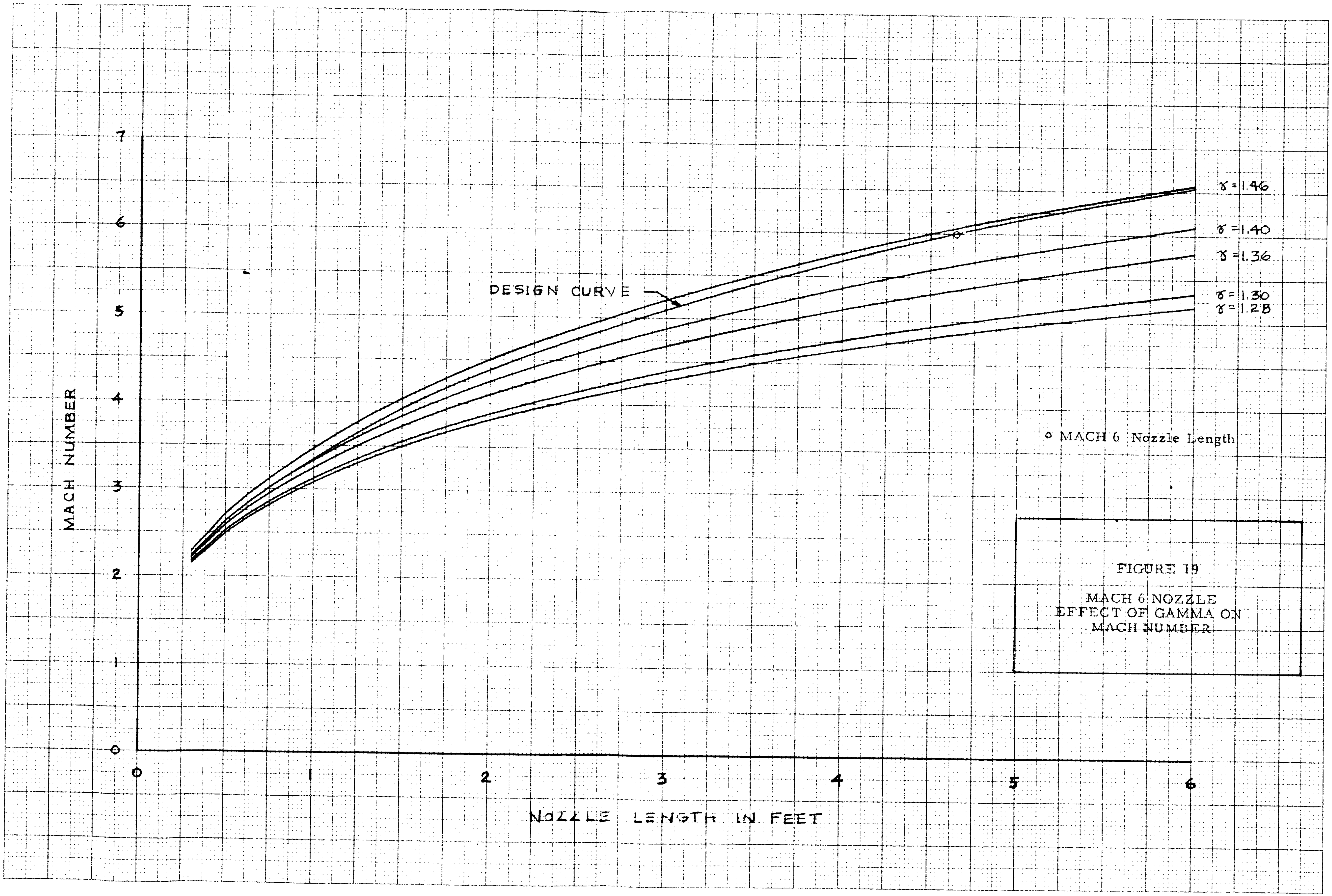
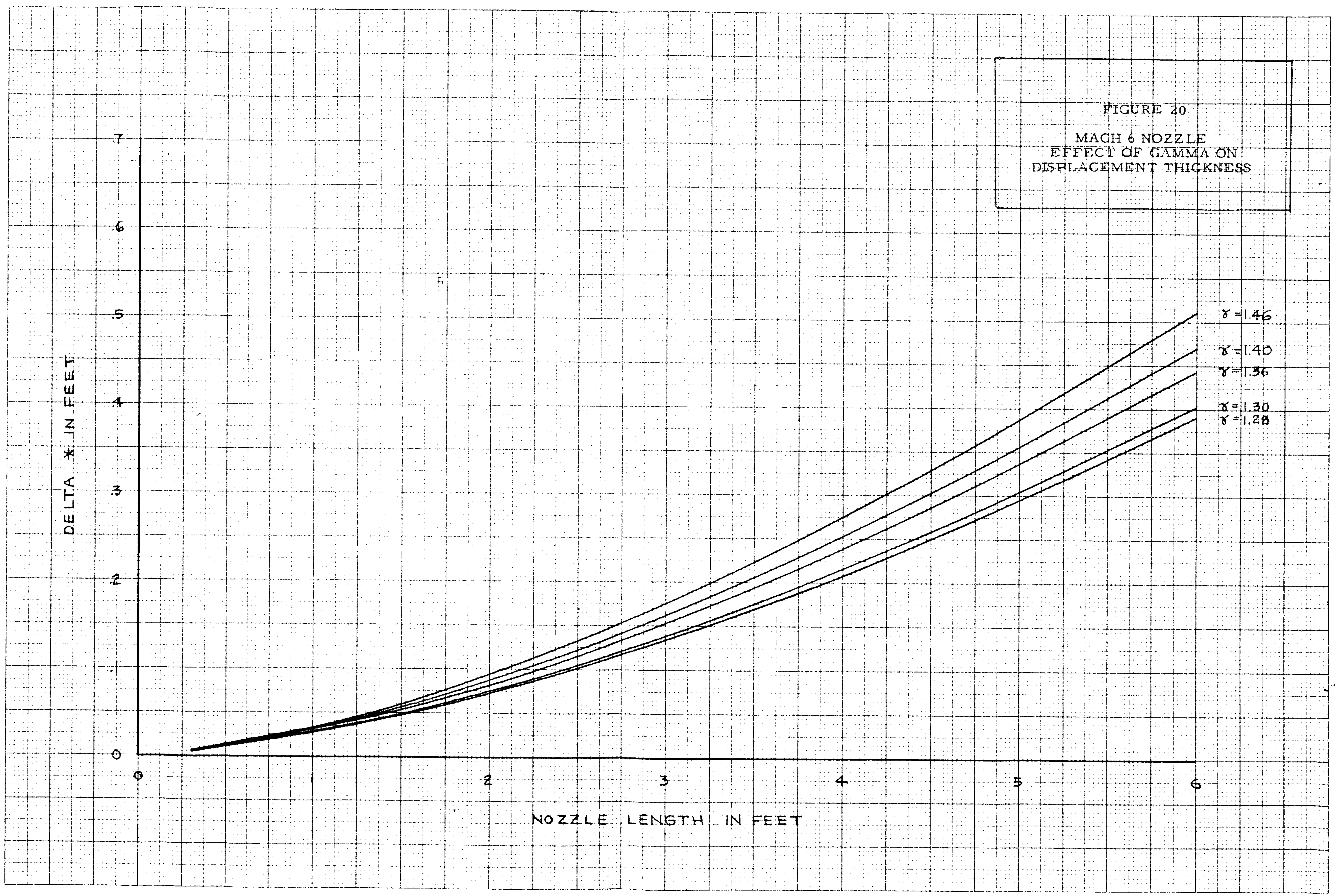


FIGURE 19

MACH 6 NOZZLE
EFFECT OF GAMMA ON
MACH NUMBER

FIGURE 20

MACH 6 NOZZLE
EFFECT OF GAMMA ON
DISPLACEMENT THICKNESS



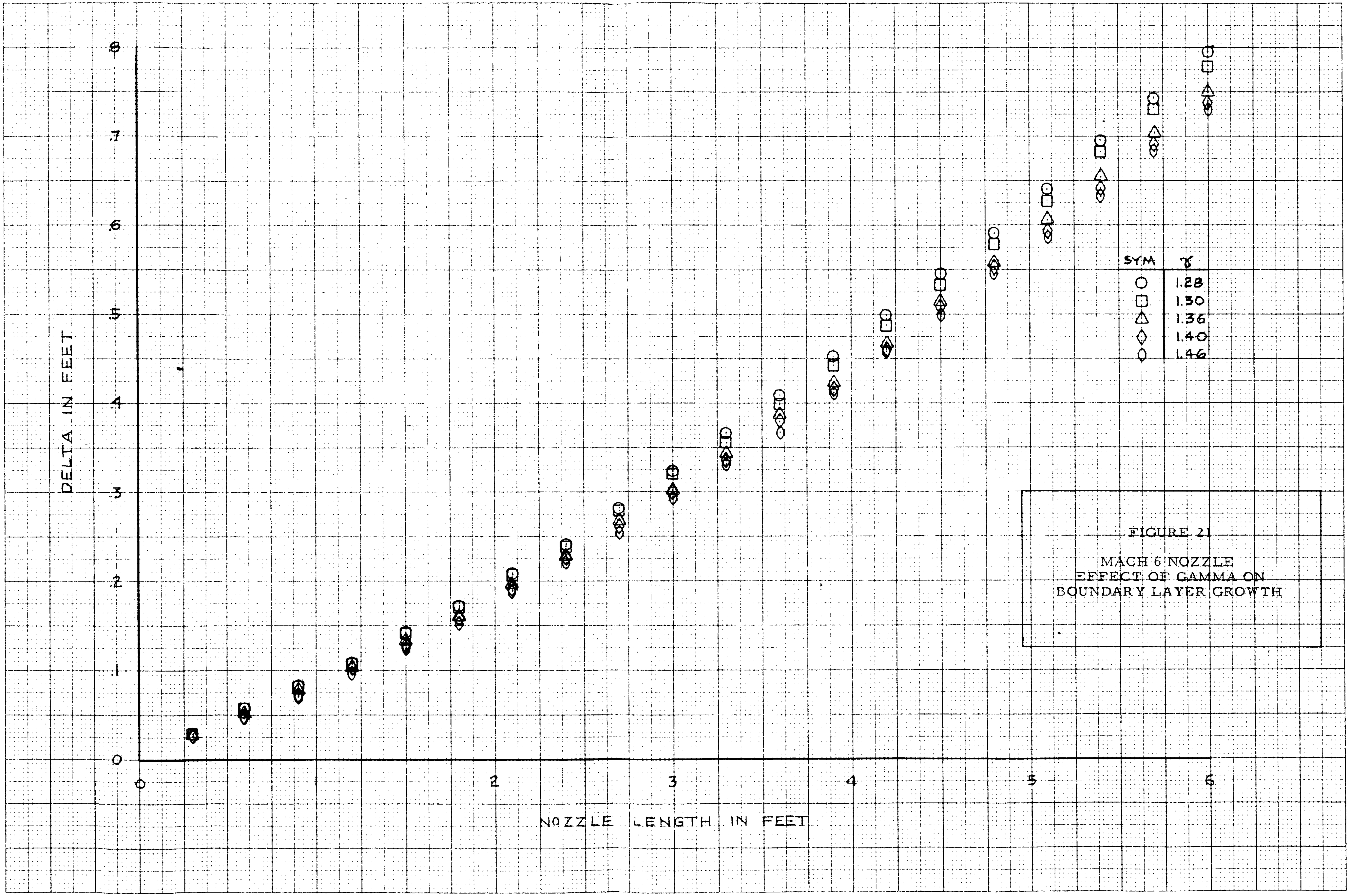


FIGURE 21

MACH 6 NOZZLE
EFFECT OF GAMMA ON
BOUNDARY LAYER GROWTH

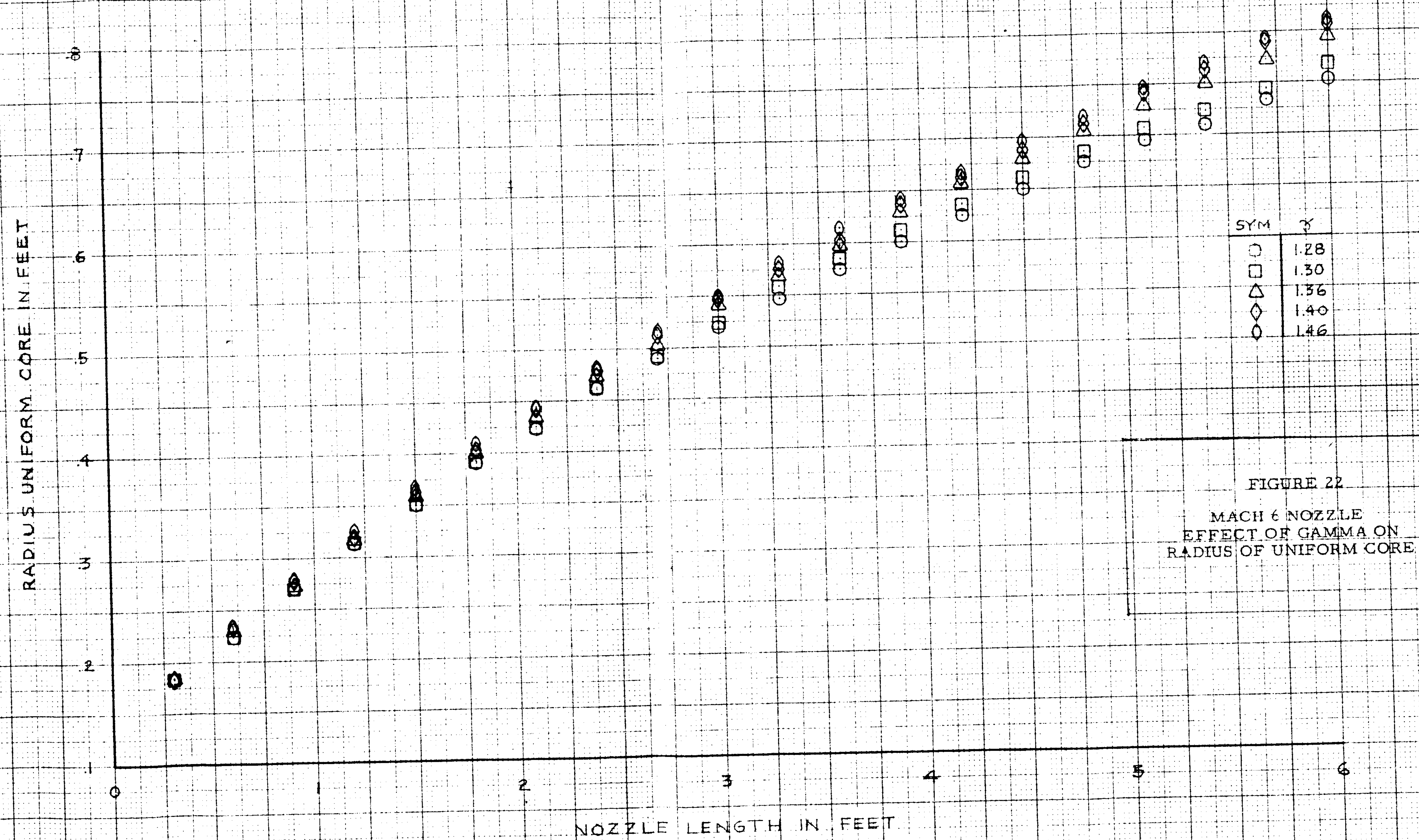


FIGURE 22

MACH 6 NOZZLE
EFFECT OF GAMMA ON
RADIUS OF UNIFORM CORE

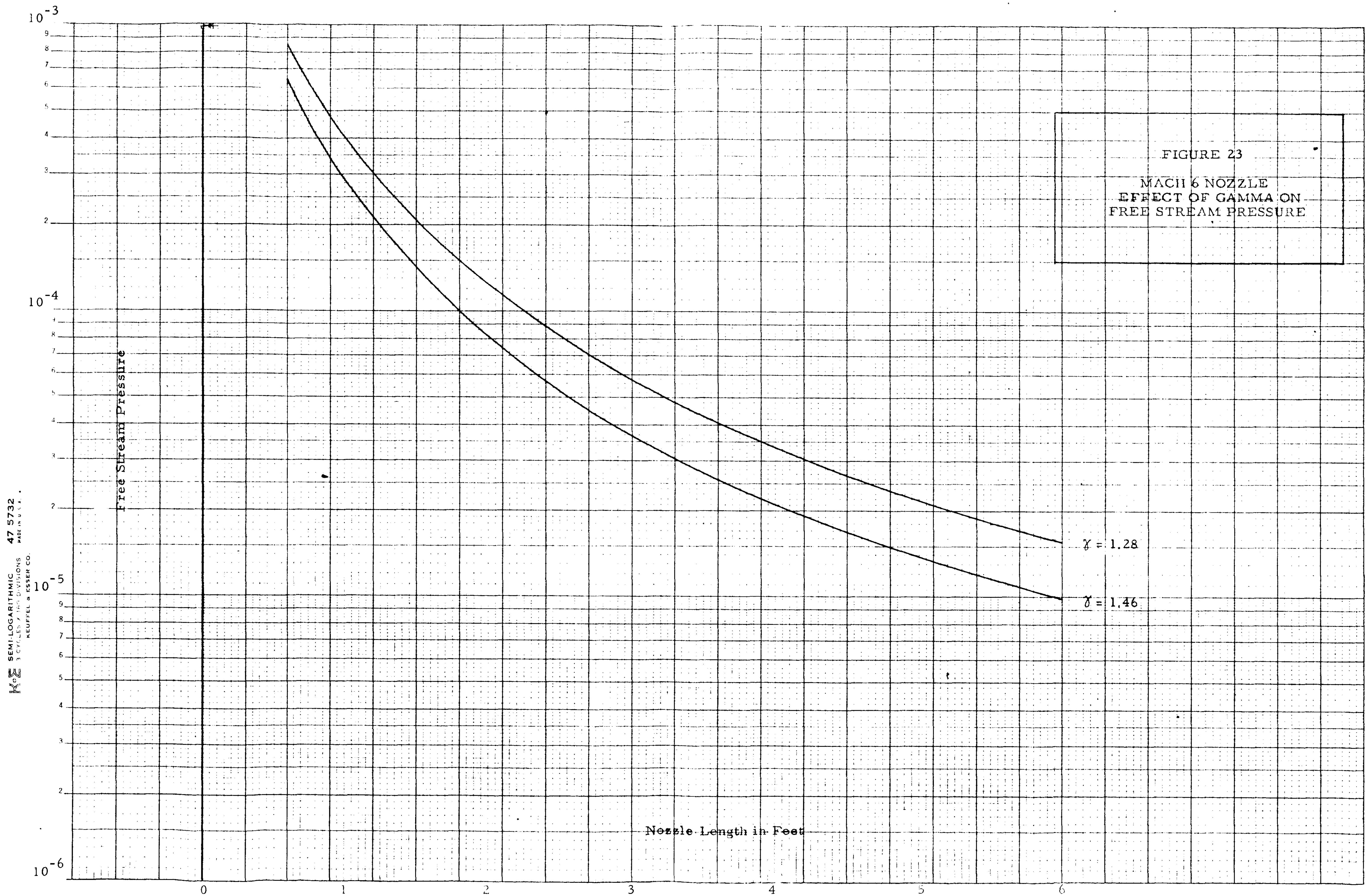


FIGURE 23
MACH 6 NOZZLE
EFFECT OF GAMMA ON
FREE STREAM PRESSURE
γ = 1.28
γ = 1.46
Free Stream Pressure
Nozzle Length in Feet
10^-3
10^-4
10^-5
10^-6
0
1
2
3
4
5
6
K+E SEMI-LOGARITHMIC 47 5732
3 CYCLES X 7TH DIVISIONS MADE IN U.S.A.
KEUFFEL & ESSER CO.

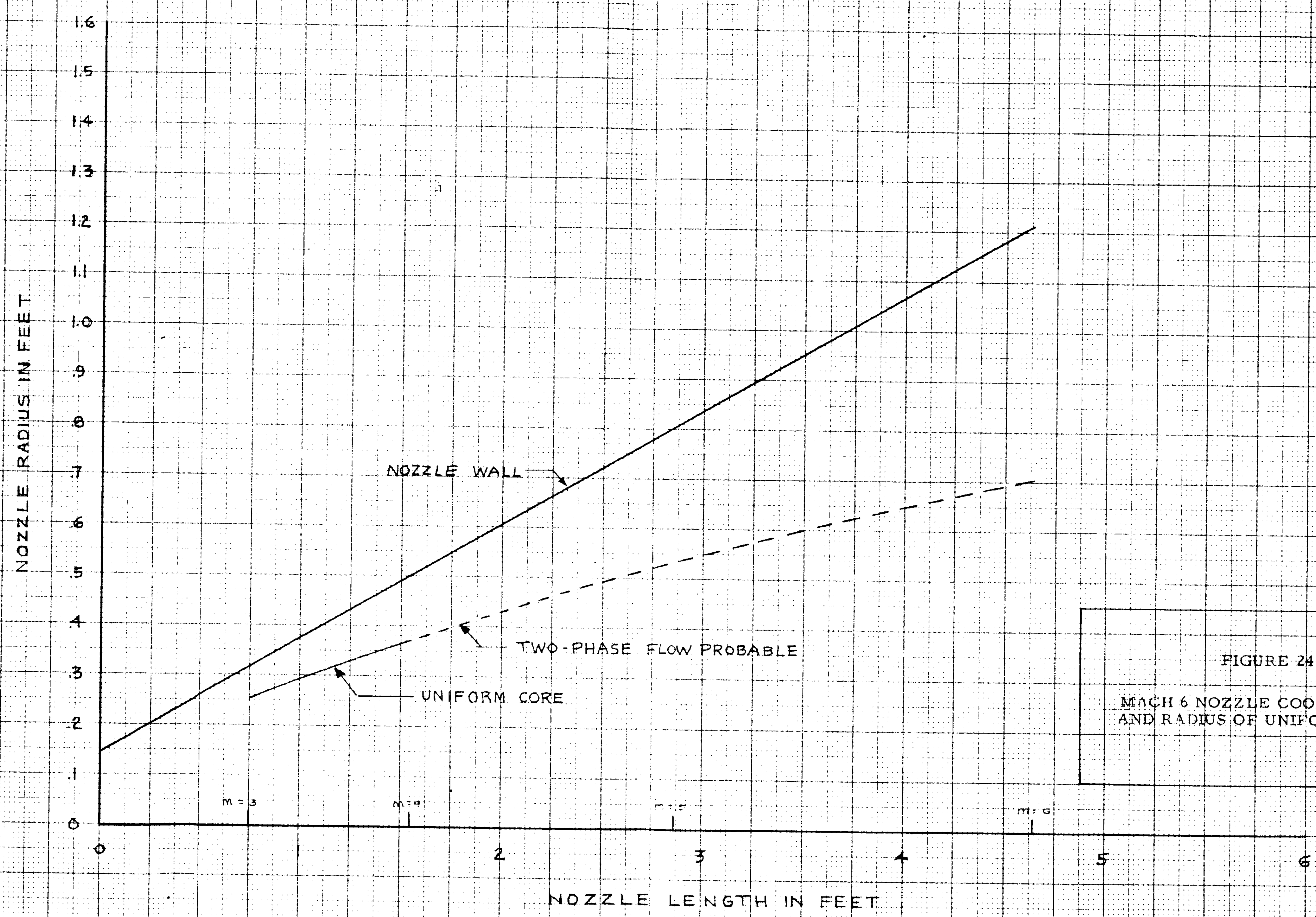


FIGURE 24

MACH 6 NOZZLE COORDINATES AND RADIUS OF UNIFORM CORE

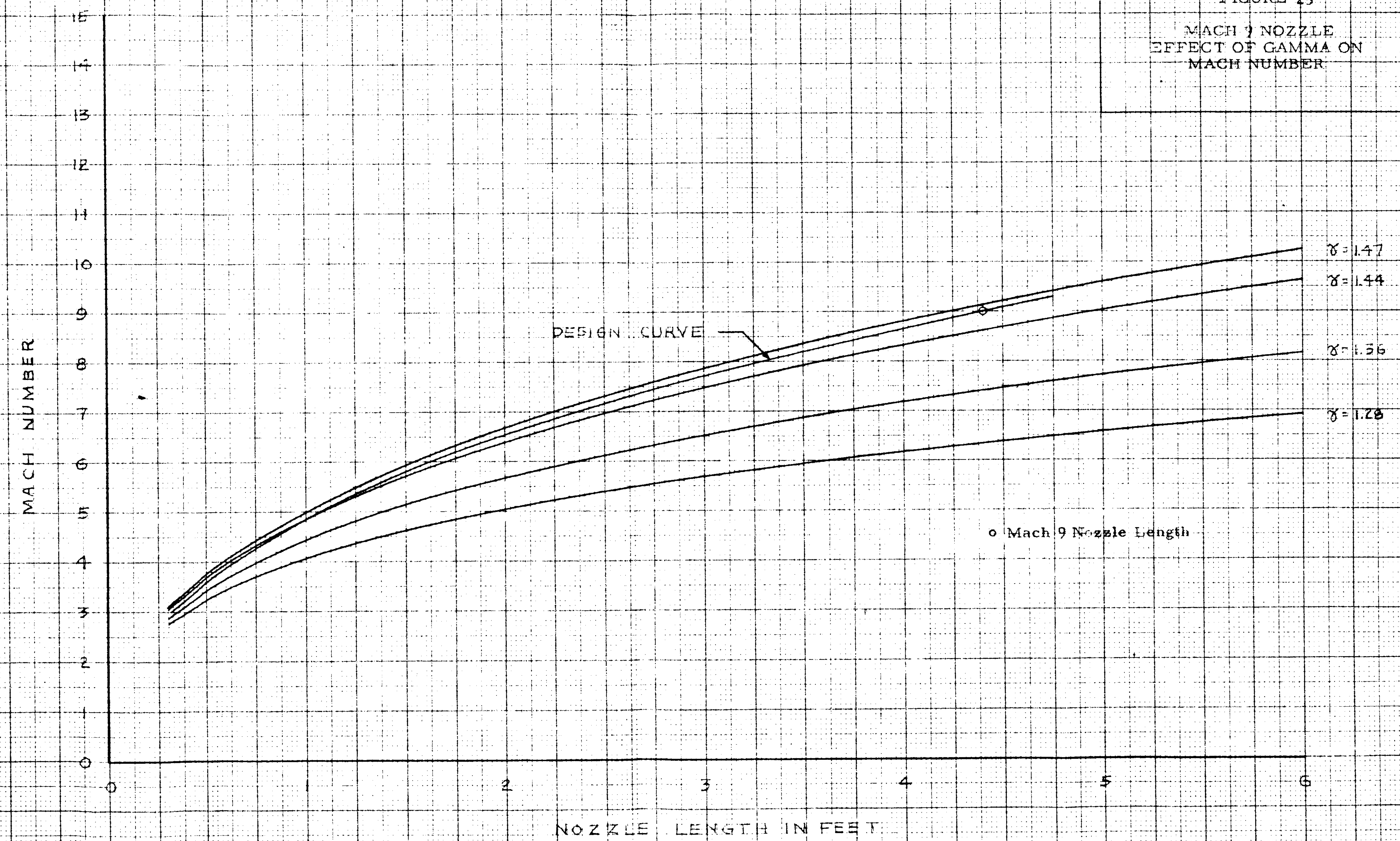
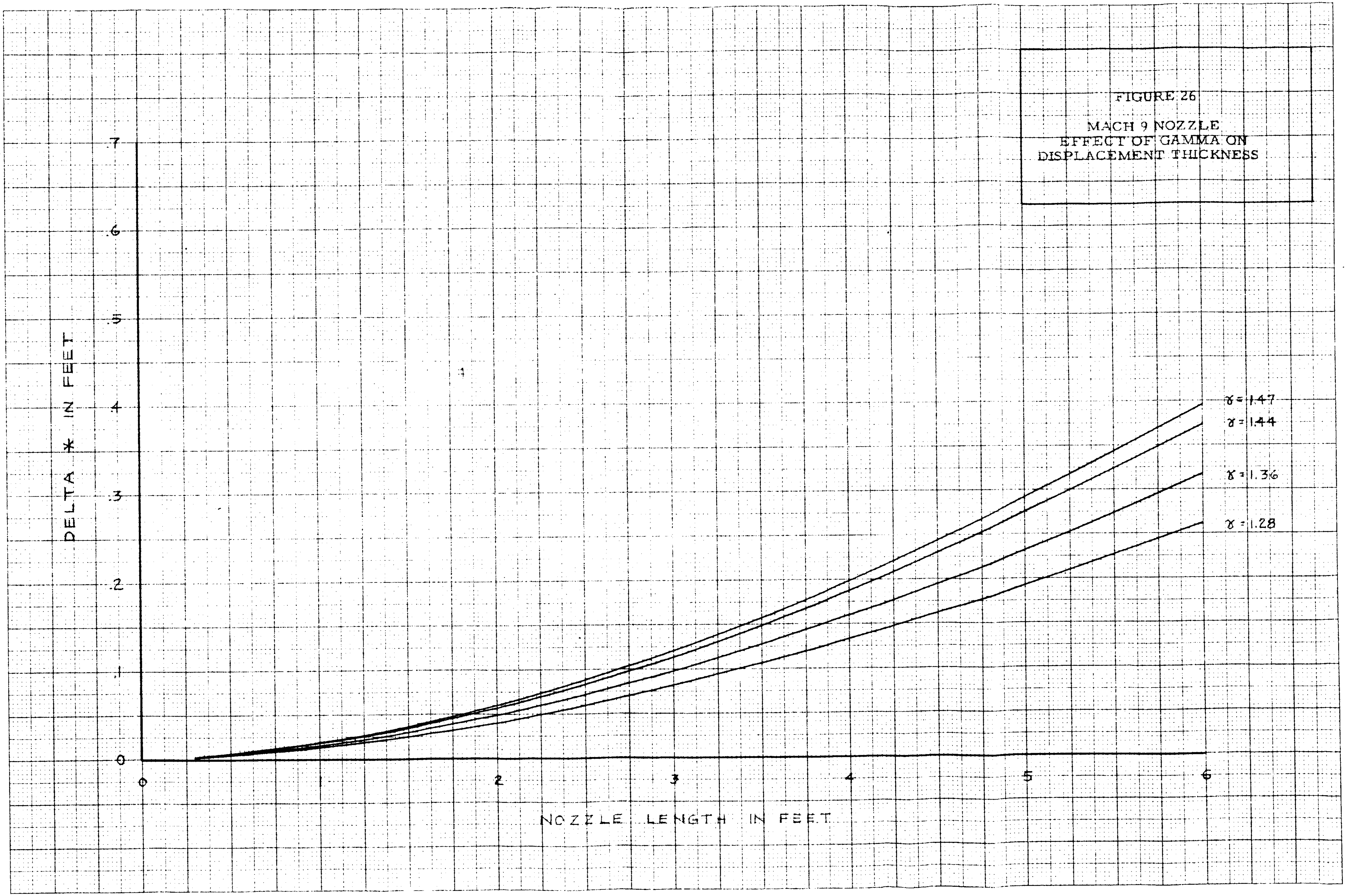


FIGURE 25

MACH 9 NOZZLE
EFFECT OF GAMMA ON
MACH NUMBER

FIGURE 26

MACH 9 NOZZLE
EFFECT OF GAMMA ON
DISPLACEMENT THICKNESS



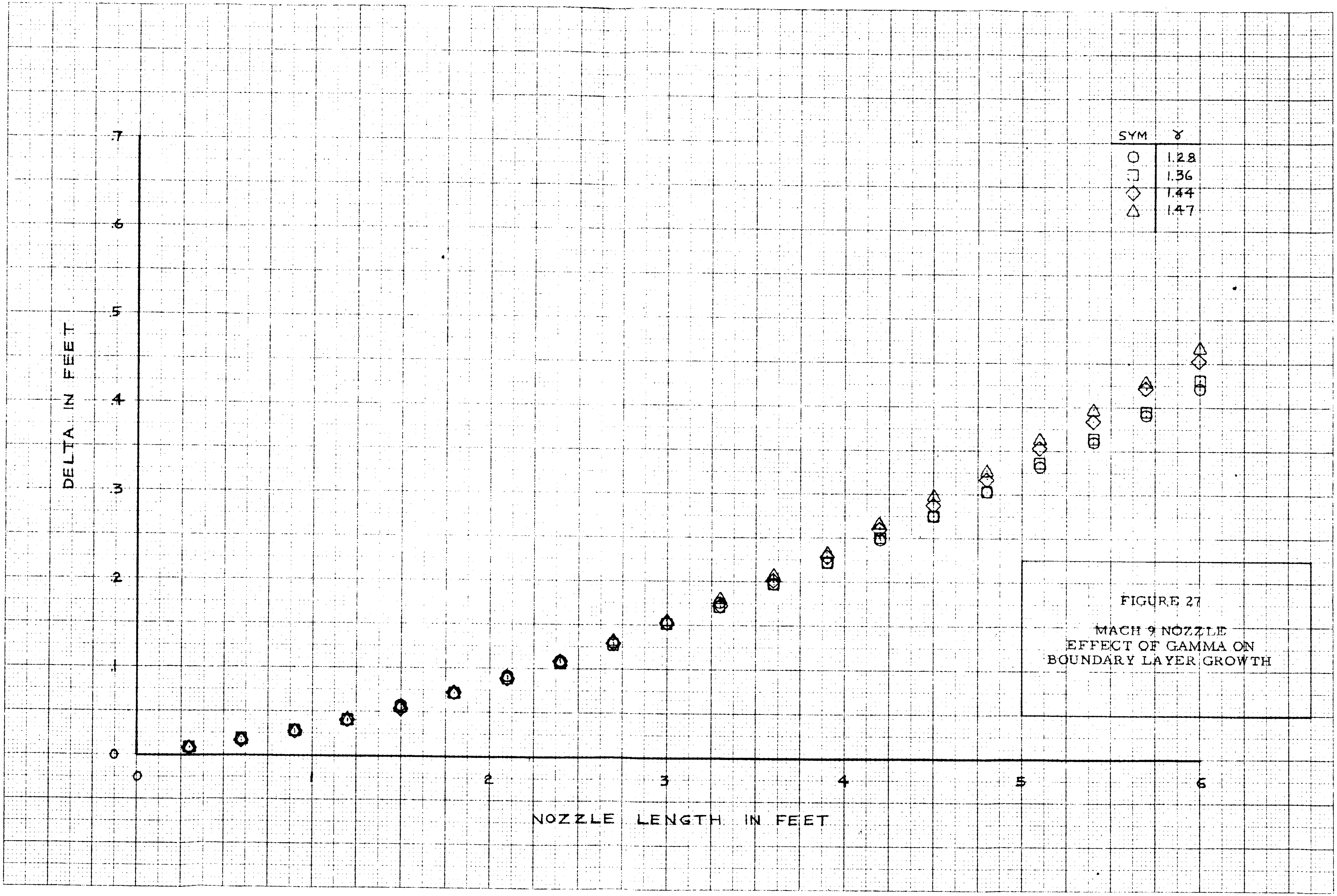


FIGURE 27

MACH 9 NOZZLE
EFFECT OF GAMMA ON
BOUNDARY LAYER GROWTH

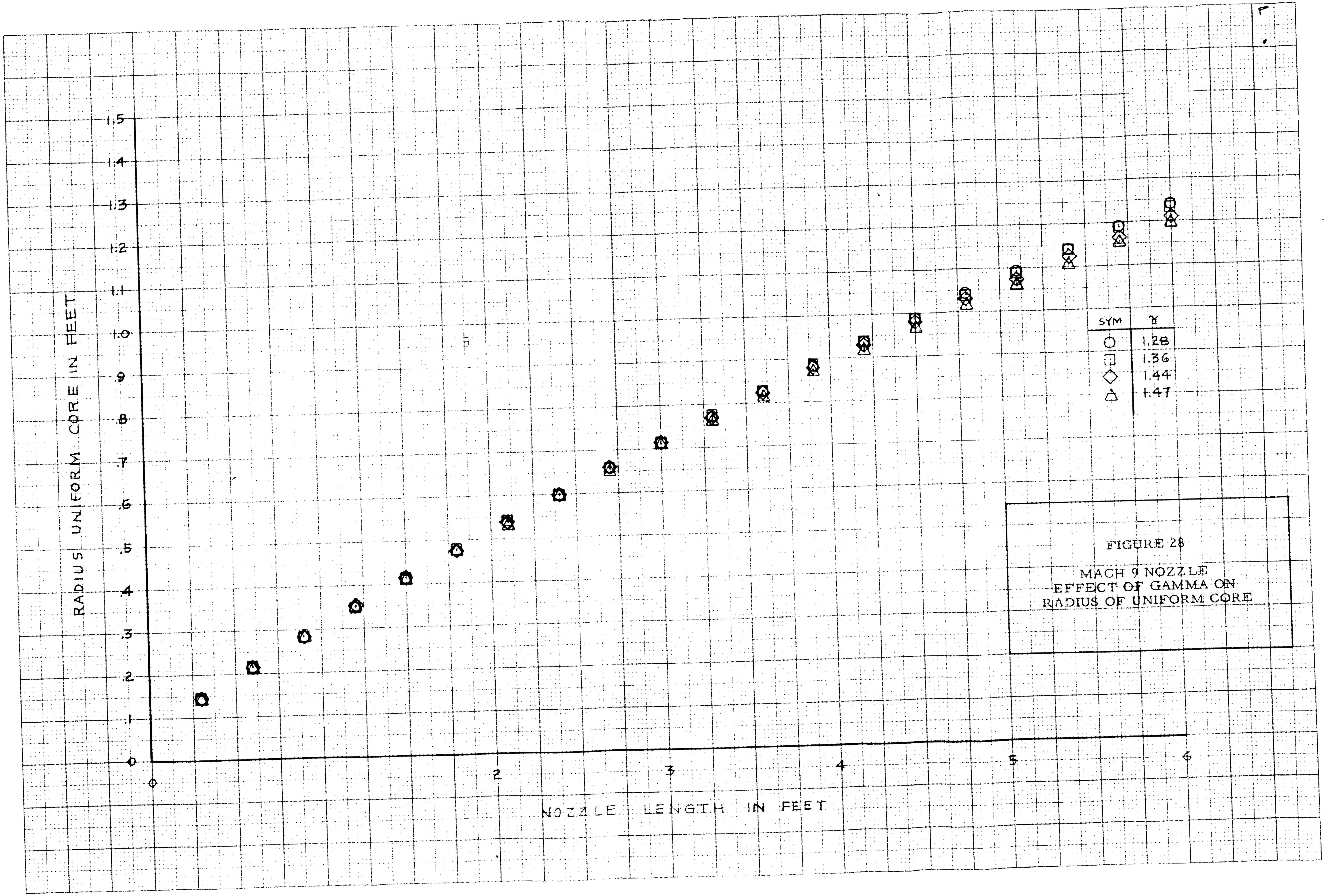


FIGURE 28

MACH 9 NOZZLE
EFFECT OF GAMMA ON
RADIUS OF UNIFORM CORE

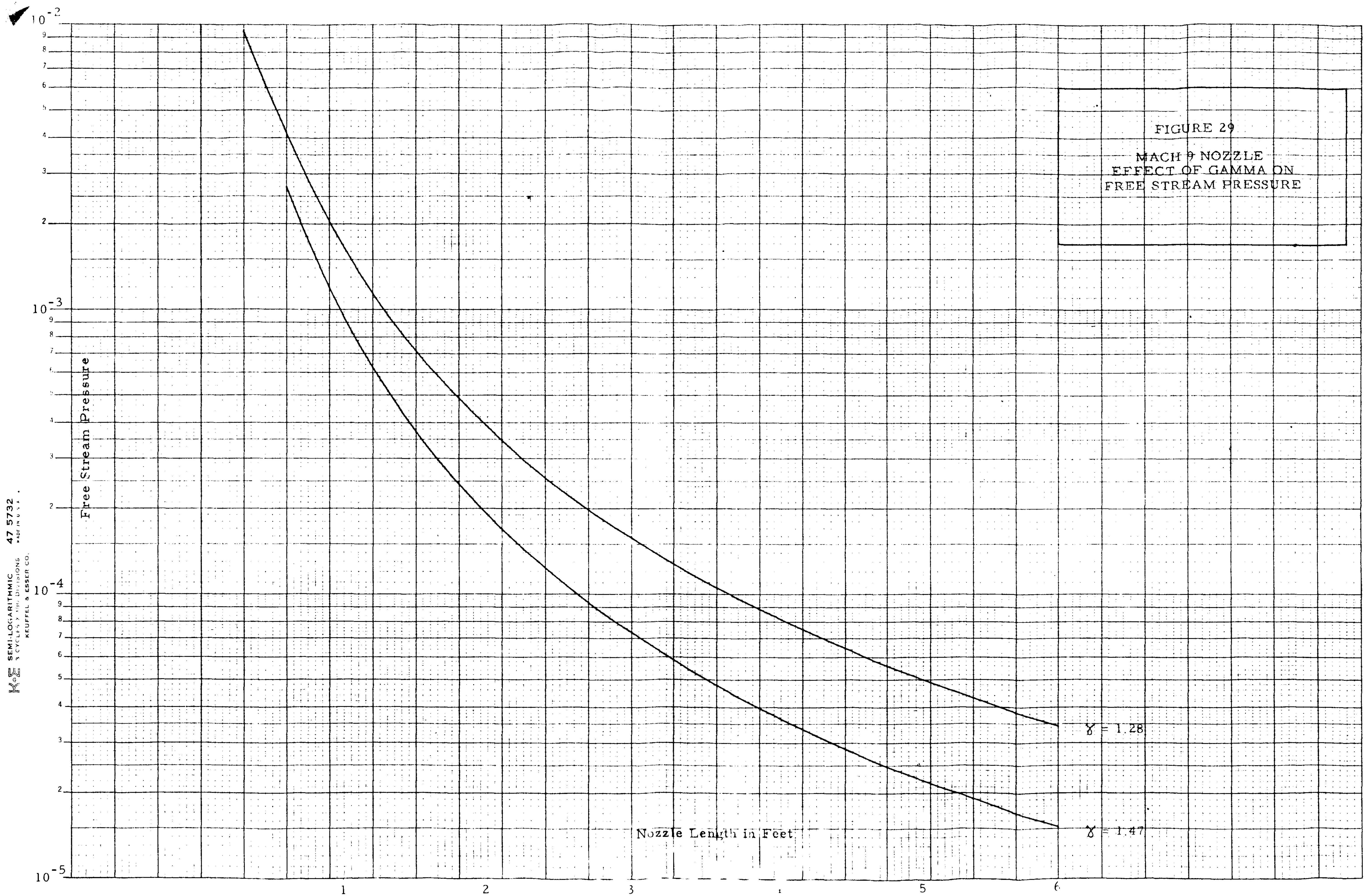


FIGURE 29
MACH 9 NOZZLE
EFFECT OF GAMMA ON
FREE STREAM PRESSURE
Free Stream Pressure
Nozzle Length in Feet
$\gamma = 1.28$
$\gamma = 1.47$
10^{-2}
10^{-3}
10^{-4}
10^{-5}
1
2
3
5
6
SEMI-LOGARITHMIC
47 5732
KEUFFEL & ESSER CO.

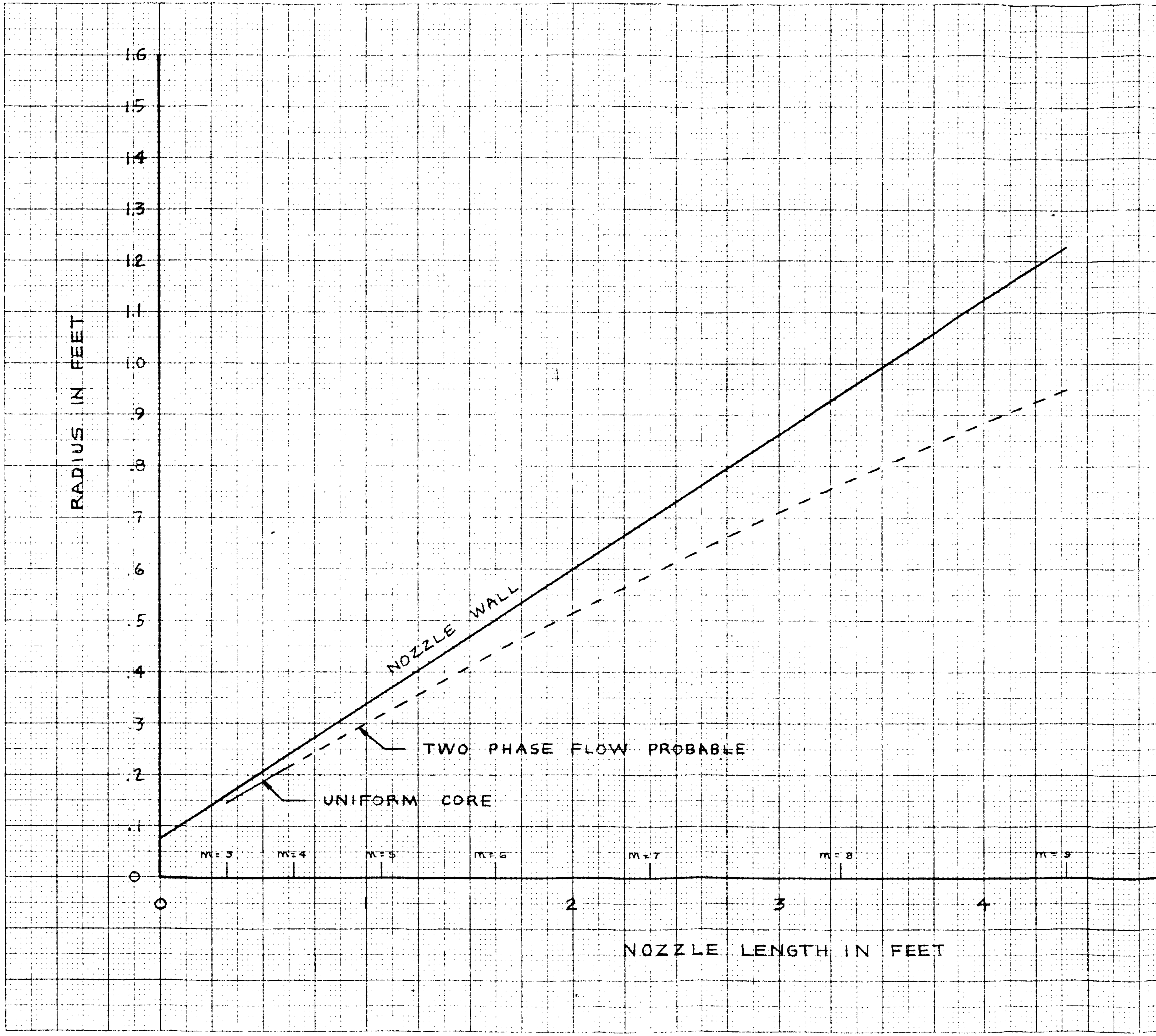


FIGURE 30

MACH 9 NOZZLE COORDINATES AND RADIUS OF UNIFORM CORE

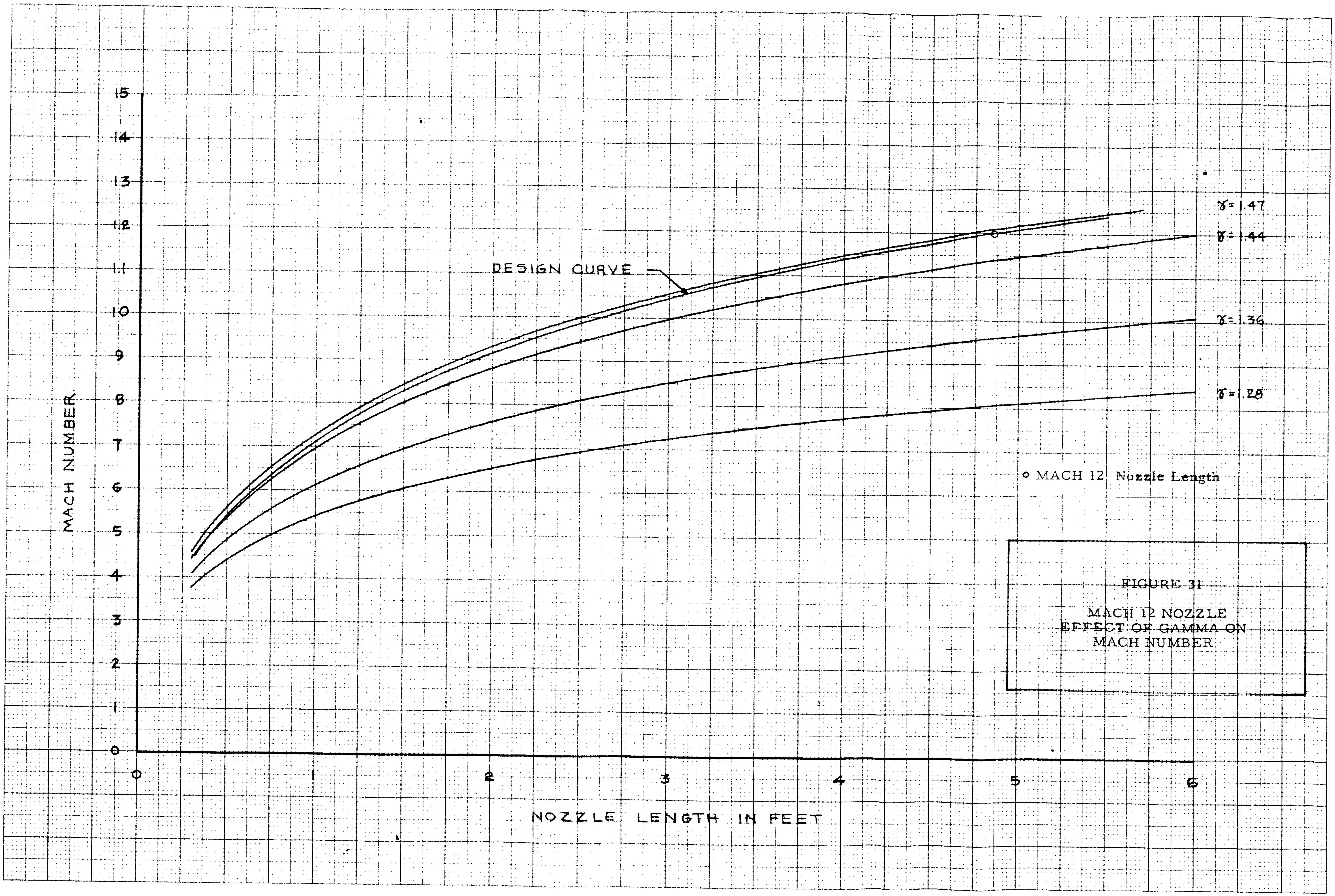


FIGURE 31

MACH 12 NOZZLE
EFFECT OF GAMMA ON
MACH NUMBER

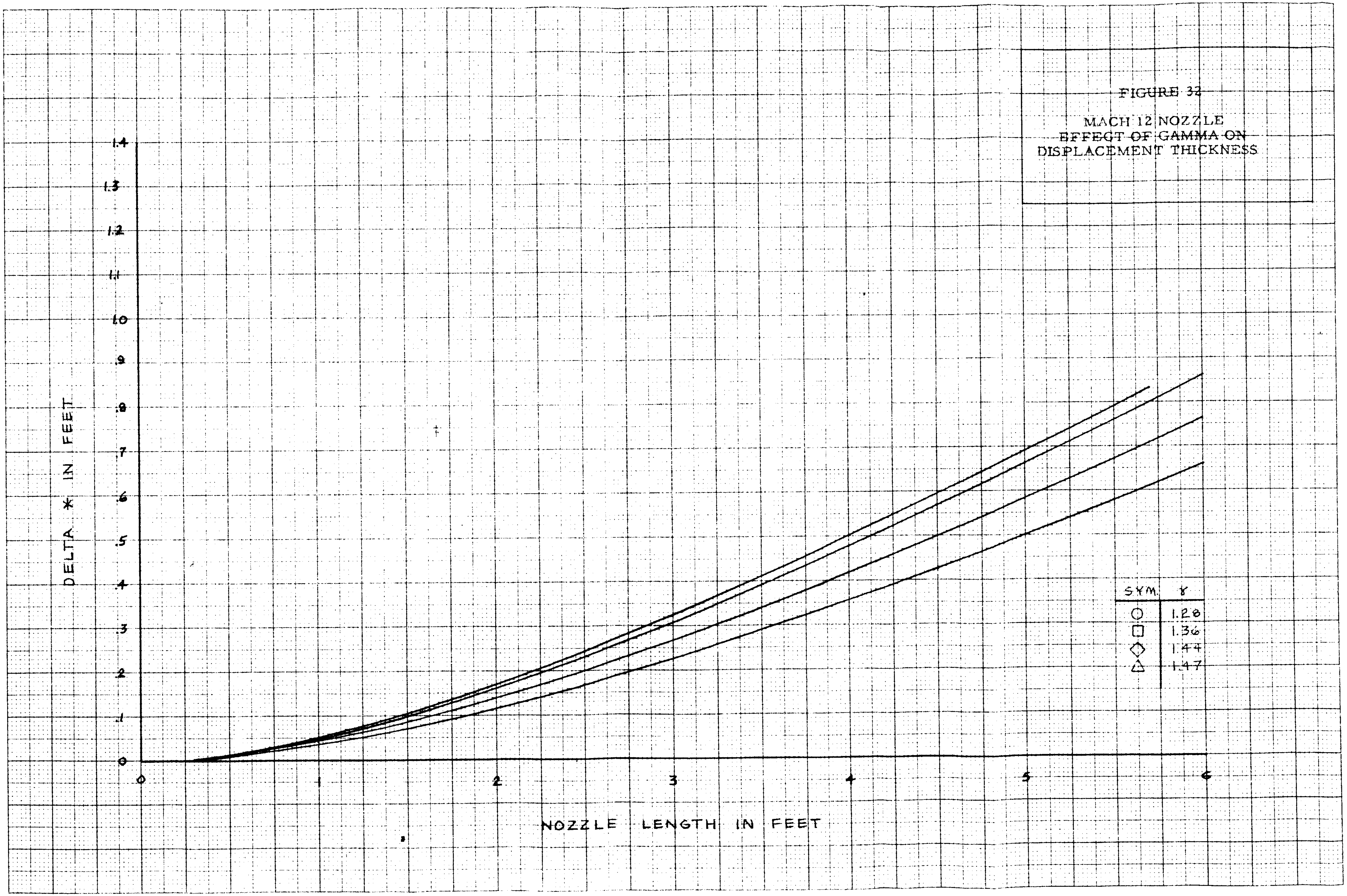


FIGURE 32
MACH 12 NOZZLE
EFFECT OF GAMMA ON
DISPLACEMENT THICKNESS
SYM.
γ
1.28
1.36
1.44
1.47
DELTA * IN FEET
0
.1
.2
.3
.4
.5
.6
.7
.8
.9
1.0
1.1
1.2
1.3
1.4
NOZZLE LENGTH IN FEET
0
1
2
3
4
5
6

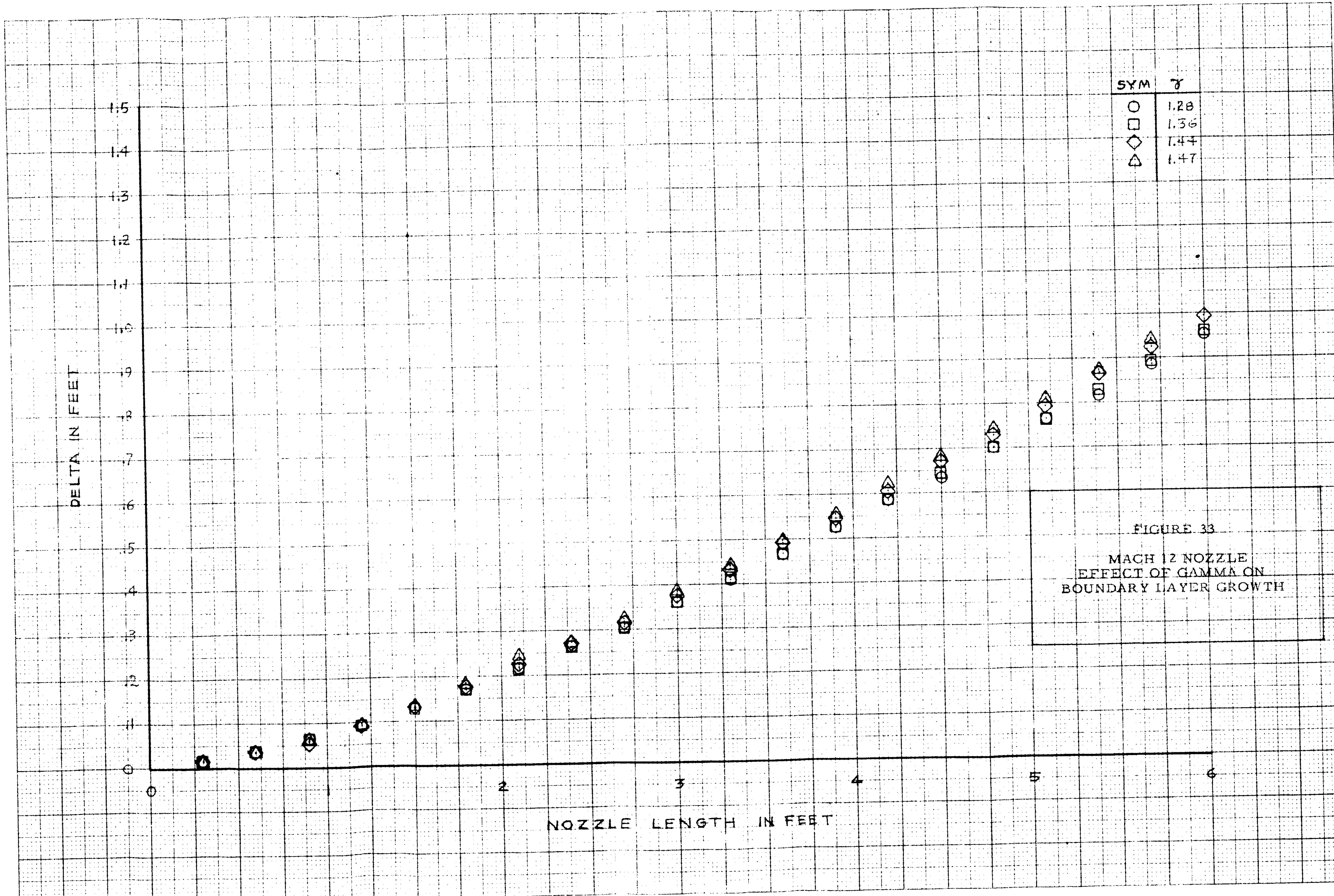


FIGURE 33

MACH 12 NOZZLE
EFFECT OF GAMMA ON
BOUNDARY LAYER GROWTH

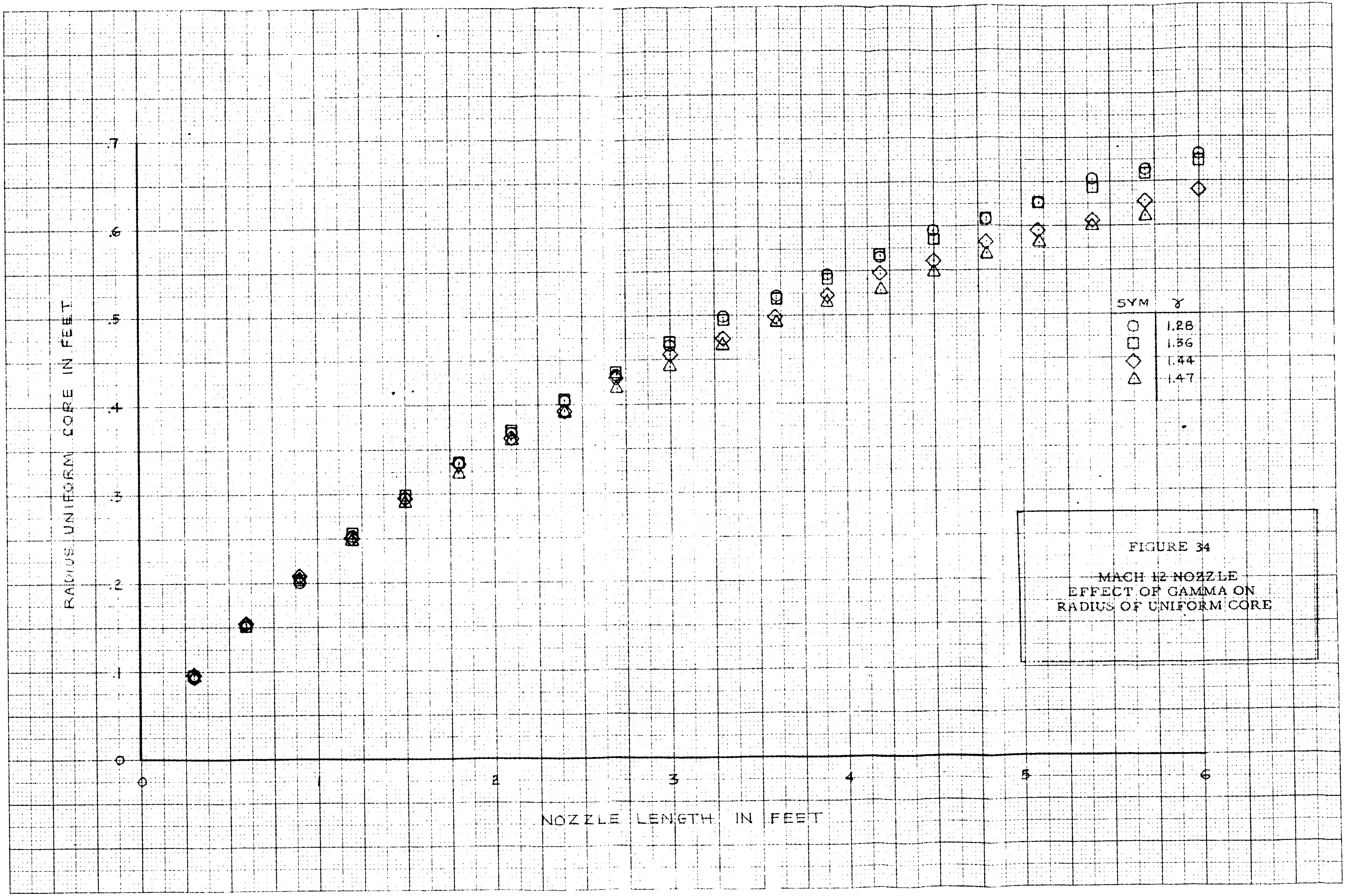


FIGURE 34

MACH 12 NOZZLE
EFFECT OF GAMMA ON
RADIUS OF UNIFORM CORE

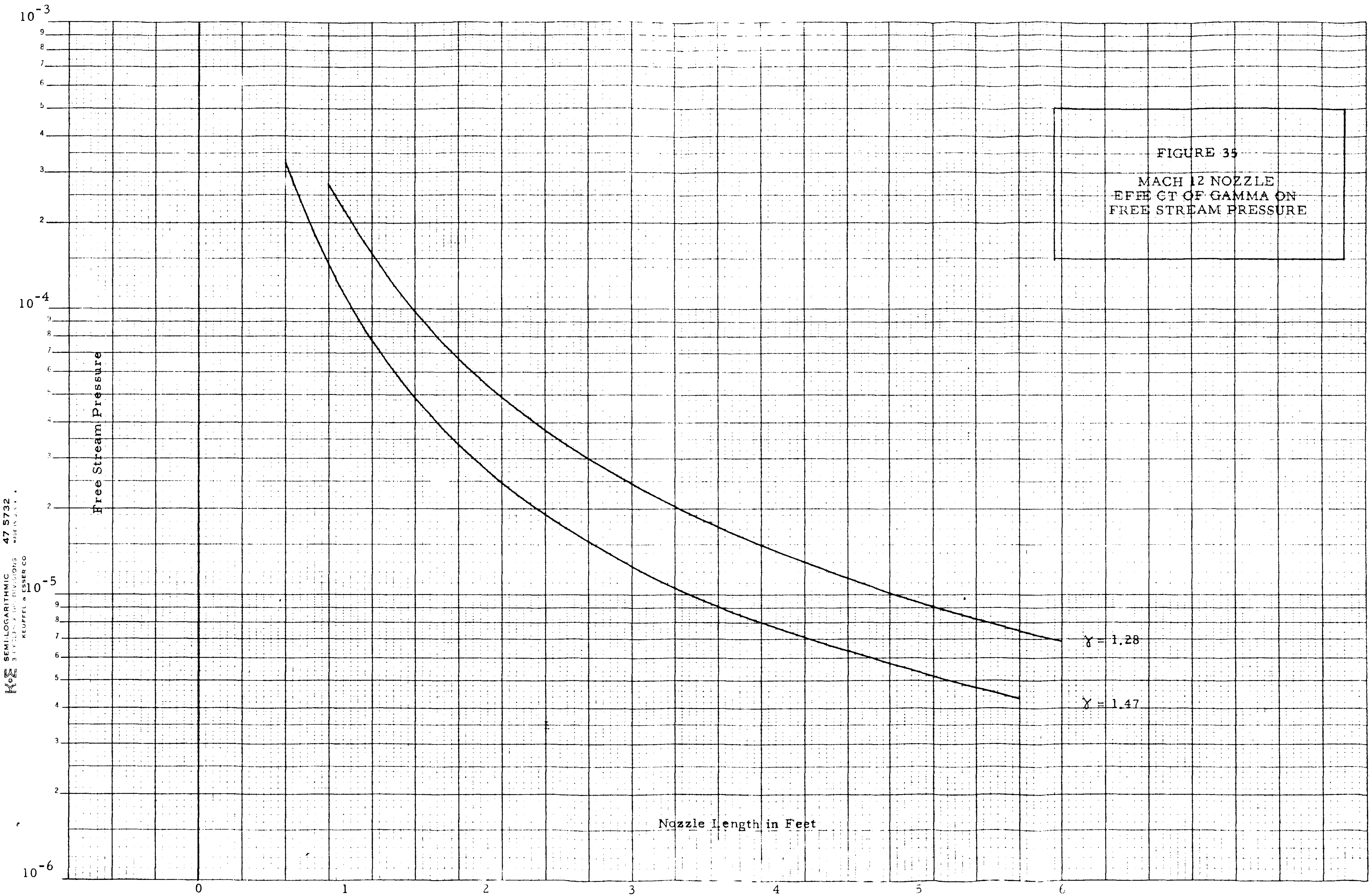


FIGURE 35
MACH 12 NOZZLE
EFFECT OF GAMMA ON
FREE STREAM PRESSURE
$\gamma = 1.28$
$\gamma = 1.47$
Free Stream Pressure
Nozzle Length in Feet
10^{-3}
10^{-4}
10^{-5}
10^{-6}
0
1
2
3
4
5
6
K&E SEMI-LOGARITHMIC 47 5732
3 CYCLES X 10 DIVISIONS
KEUFFEL & ESSER CO

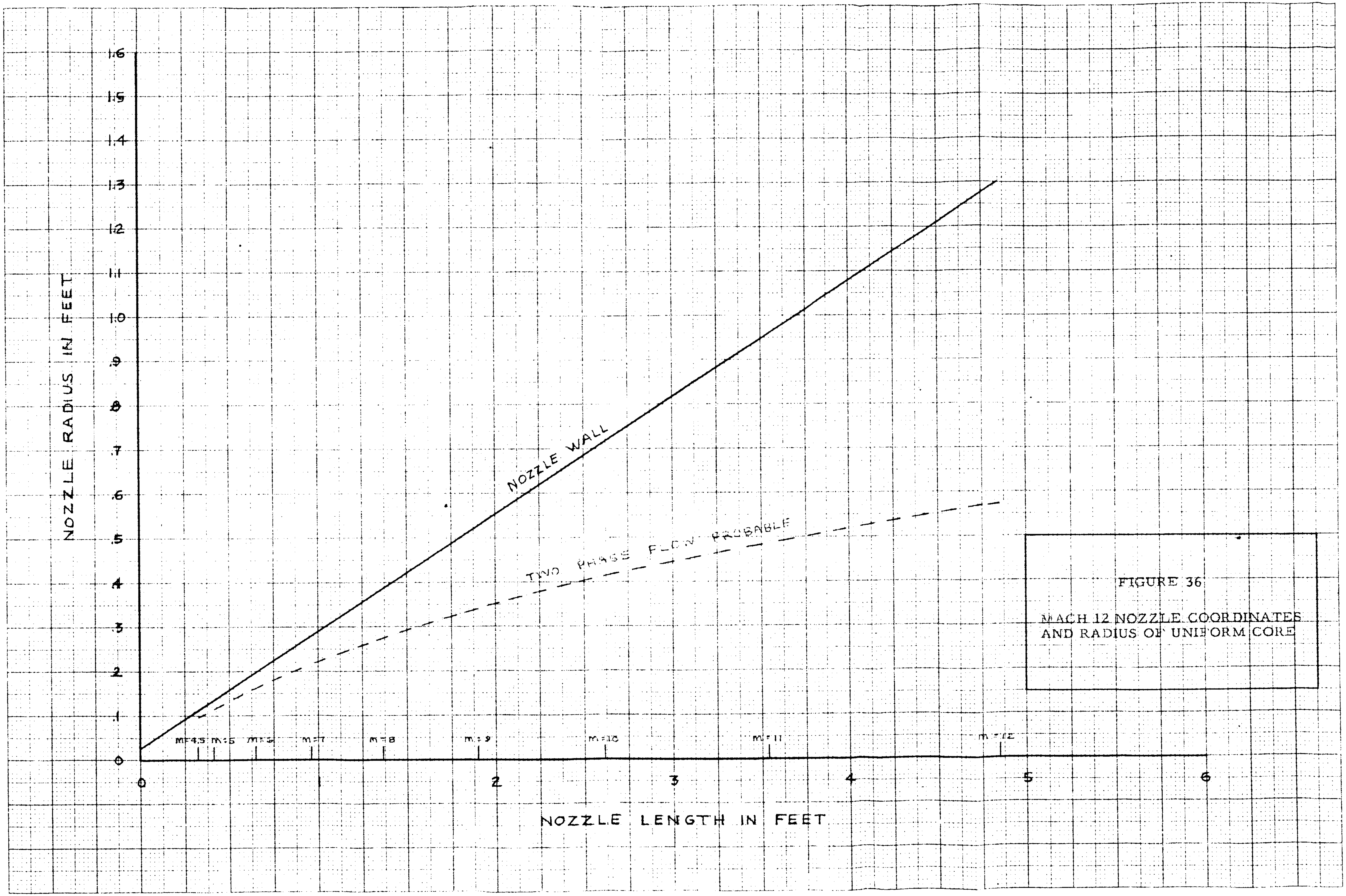


FIGURE 36

MACH 12 NOZZLE COORDINATES
AND RADIUS OF UNIFORM CORE

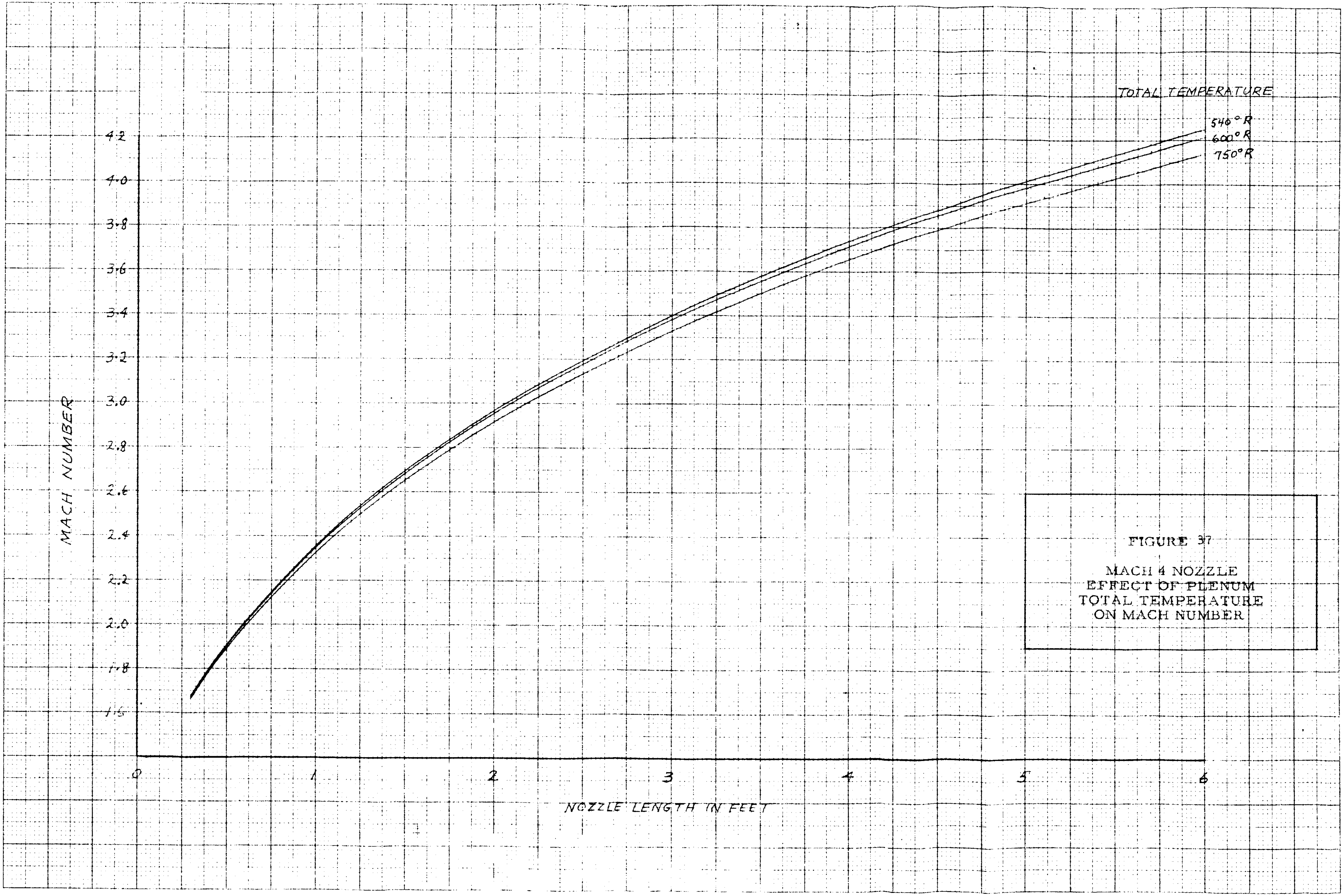


FIGURE 37

MACH 4 NOZZLE
EFFECT OF PLENUM
TOTAL TEMPERATURE
ON MACH NUMBER

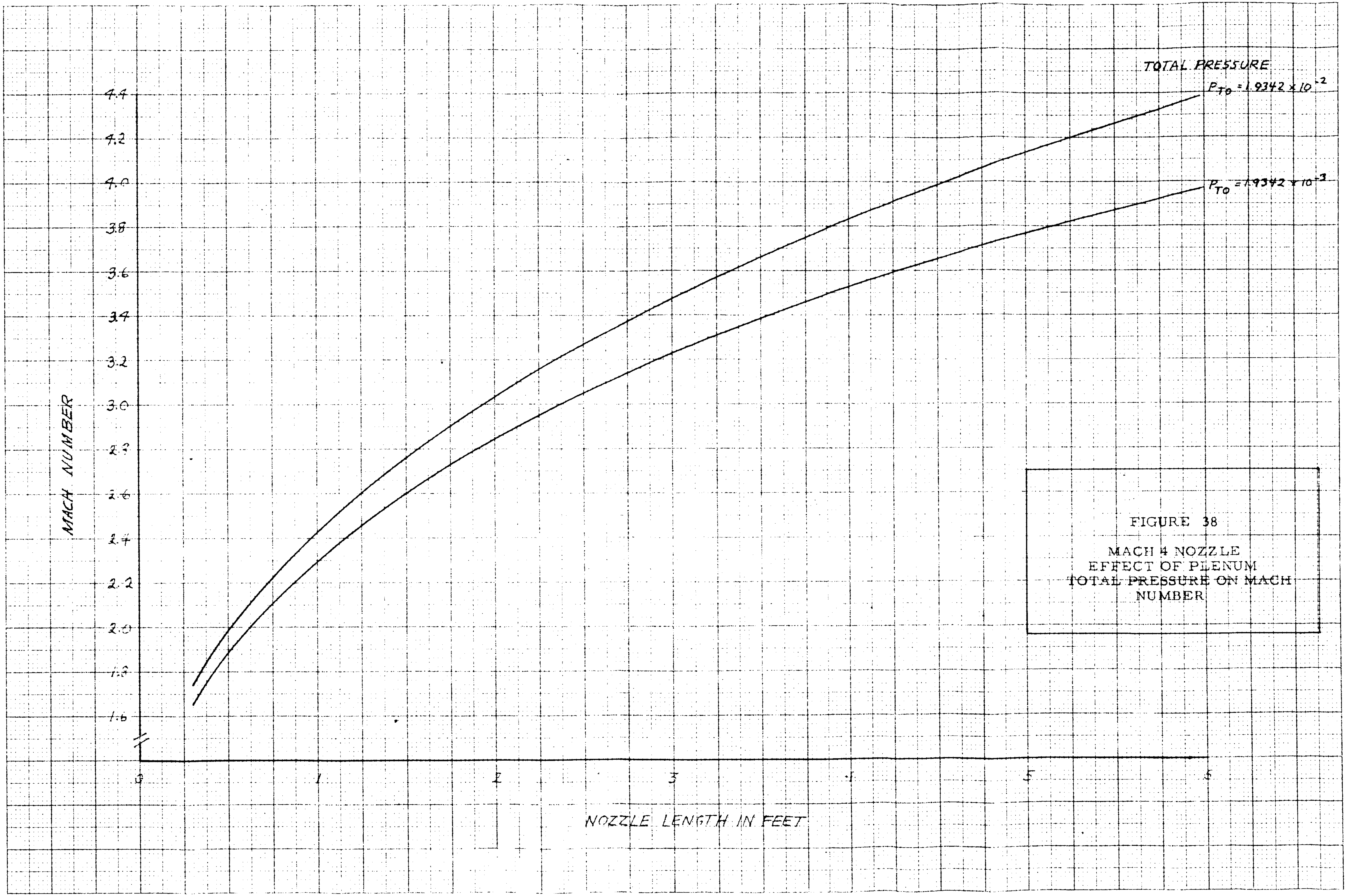


FIGURE 38

MACH 4 NOZZLE
EFFECT OF PLENUM
TOTAL PRESSURE ON MACH
NUMBER

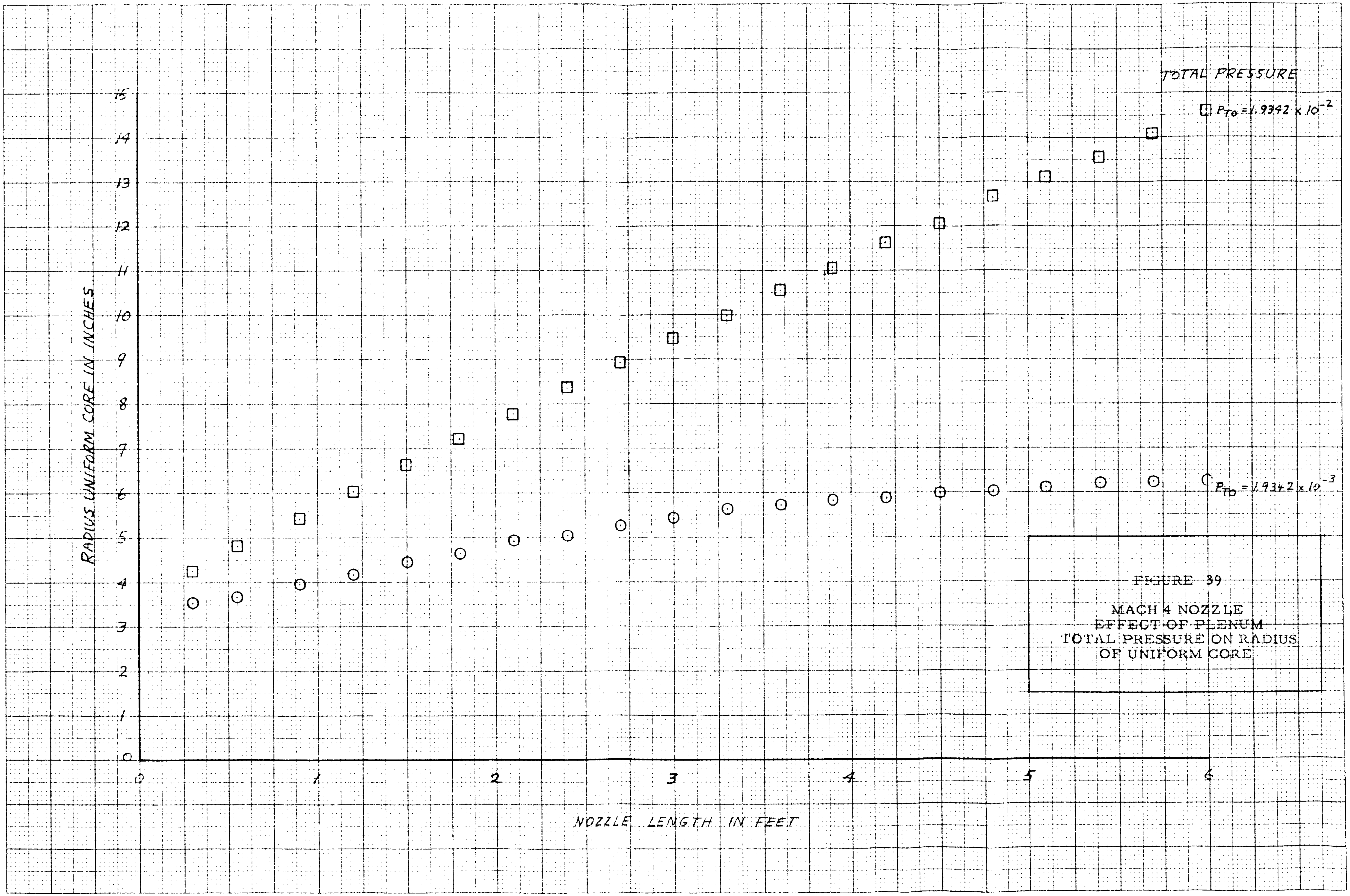


FIGURE 39

MACH 4 NOZZLE
EFFECT OF PLENUM
TOTAL PRESSURE ON RADIUS
OF UNIFORM CORE

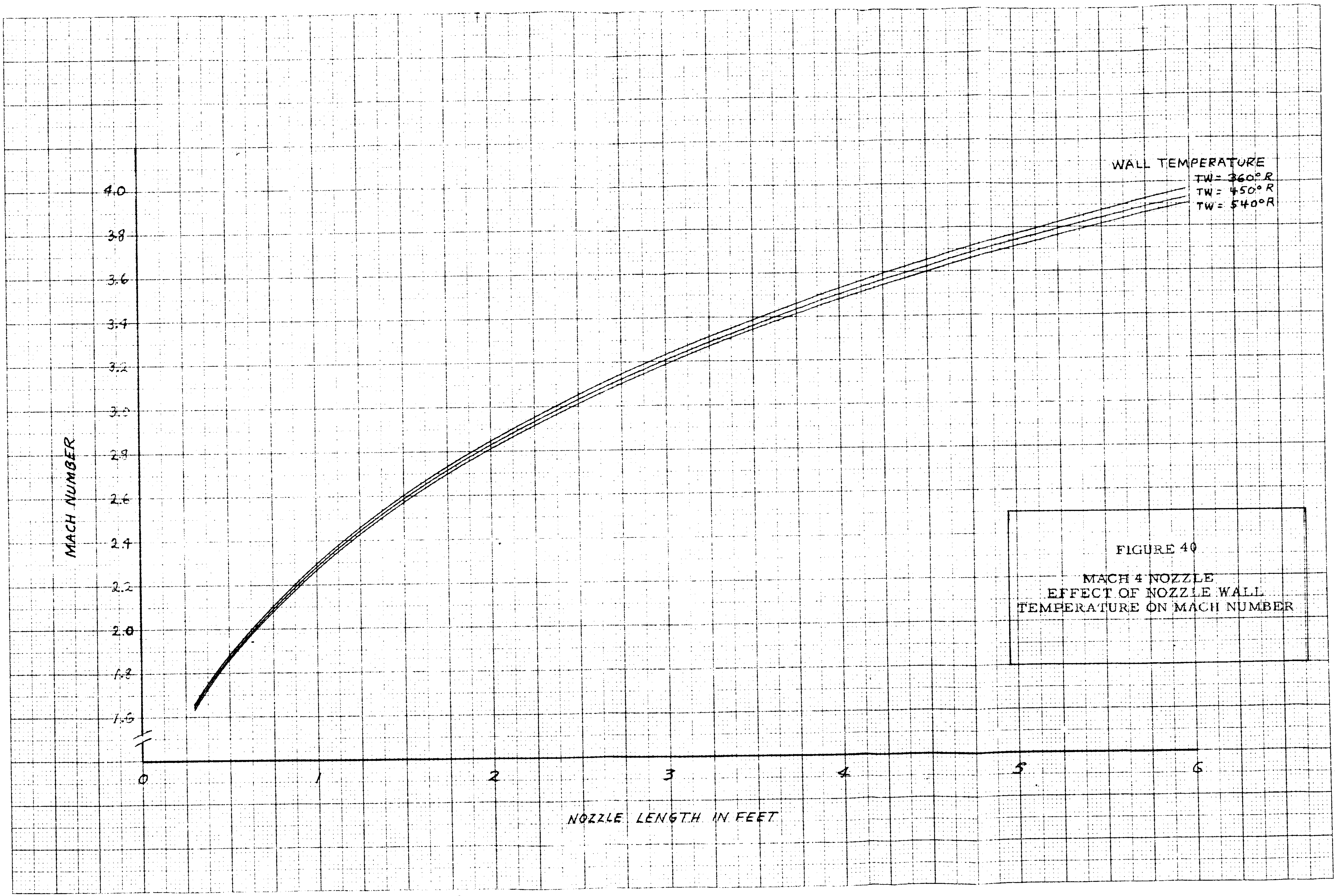


FIGURE 40

MACH 4 NOZZLE
EFFECT OF NOZZLE WALL
TEMPERATURE ON MACH NUMBER